CHAPTER 2. PART 29
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

SUBPART E - POWERPLANT

POWERPLANT - GENERAL

AC 29.901. § 29.901 (Amendment 29-17) INSTALLATION.

a. Section 29.901(a):

(1) Explanation. Paragraph (a) provides a definition of areas of rotorcraft for which safety requirements are set forth under the general title, SUBPART E - POWERPLANT. This subpart includes not only major propulsive elements and power transmissive components but also powerplant controls and instruments, safety devices, including fire protection and other devices to protect personnel, and critical flight structure in event of fires.

(2) Procedures. To ensure that no certification aspect is overlooked in establishing compliance, certification engineers should make at least an informal breakdown of all components of the rotorcraft, assigning responsibility to powerplant certification engineers of all items within the above definition. While this procedure is usually straightforward, the following items of FAA/AUTHORITY powerplant responsibility are listed to minimize questions regarding authority and responsibility.

(i) Drive system components. All parts of the transmission, clutches, shafting, including the driveshafts (masts) of main and auxiliary rotors, powerplant cooling components, and powerplant instrumentation requirements under §§ 29.1305, 29.1337, 29.1543, 29.1549, 29.1551, 29.1553, 29.1555, and 29.1583.

NOTE: The division of responsibility between FAA/AUTHORITY airframe engineers and FAA/AUTHORITY powerplant engineers (in accordance with FAA/AUTHORITY practice) regarding the driveshaft is at the flange or spline interface between the driveshaft and the rotor hub. Rotor hubs, controls, blades, and associated components are the airframe engineers' responsibility. (Industry practice may not agree with this concept.)

(ii) Engines, except for mount structure.

(iii) Auxiliary power units, except for mount structure.

(iv) Combustion heaters, except for downstream ventilation air ducting, mixing, and distribution systems and for electrical aspects of controls and safety devices.

(v) Water/alcohol or other fluid power augmentation systems.

(vi) Engine induction systems including induction icing and snow ingestion, and exhaust systems, including exhaust shrouds and drains.

(vii) All fuel systems, including those serving engines, auxiliary power units, combustion heaters, power augmentation systems, etc., and vents and drains for those systems.

(viii) Oil systems for engines, auxiliary power units, rotor drive transmissions, and gearboxes, including grease lubricated gears and bearings of the drive system.

(ix) Cooling aspects of engines, rotordrive transmissions and gearboxes, and auxiliary power units (APU). Electrical generating equipment and hydraulic component cooling may be the responsibility of the systems and equipment engineer provided agreement is established among responsible personnel.

(x) Rotor brakes, except hydraulic, electrical, and structural aspects of nonrotating brake components.

(xi) Fire protection, including firewalls, fire extinguisher systems, fire detector systems, flammable fluid lines, fittings, and shutoff valves. The powerplant engineer has responsibility for evaluating compliance with §§ 29.861 and 29.863 as they pertain to fuel and oil systems.

(xii) Engine and transmission cowling and covering, including latches.

(xiii) Powerplant flexible controls (reference § 29.1141(c)).

(xiv) Powerplant accessories.

(xv) Pneumatic systems (engine bleed air) within the engine or APU compartments, including shut-off valves and engine isolation features of bleed systems.

(xvi) Powerplant aspects of instrument markings and powerplant aspects of flight manuals, including limitations, normal and emergency procedures, engine performance; powerplant aspects of maintenance manuals, with emphasis on the limitations section of the manual and verification of the limitations established under § 29.1521.

b. <u>Section 29.901(b)</u>:

(1) <u>Explanation</u>. Paragraph (b) requires compliance with the engine manufacturers' approved installation instructions and any applicable provisions of this subpart that the powerplant installation must be installed in a manner to ensure

continued safe operation, that accessibility for inspection and maintenance is provided, that appropriate electrical connections (ground connections) are provided, and that allowance is provided for thermal expansion of turbine engines.

(2) Procedures.

(i) Engine Installation. Compliance with most of the detail requirements in the engine installation manual can be established by test or by design features and arrangements negotiated between the rotorcraft manufacturer and the FAA/AUTHORITY powerplant engineer. Some aspects, usually involving inlet and/or exhaust distortion limitations, vibration limitations, and aircraft/engine interface items may require direct assistance and information from the engine manufacturer to determine that compliance with the installation manual exists. Fuel control/engine/rotor system torsional matching is usually a developmental problem to be worked out before presentation of the rotorcraft to the FAA/AUTHORITY; however, final flight tests for surge or stall, torsional stability, and acceleration/deceleration schedules may require direct coordination among FAA/AUTHORITY installation engineers, engine manufacturers' representatives, and the FAA/AUTHORITY engine certification engineers. These items are addressed specifically under § 29.939. Reciprocating, carburetor-equipped engines usually require a particular carburetor configuration to achieve adequate engine cooling. This configuration, identified as a "carburetor parts list," must be approved for the engine under Part 33 and should be listed with the engine on the type data sheet for the rotorcraft.

(ii) Arrangement and Construction. Each item of the powerplant area of responsibility should be shown to be suitable for its intended purpose and installed to operate satisfactorily and safely between normal inspections and overhauls. Accessories mounted on engine or transmission drive pads should be determined to be compatible with the pad limits including fit and speed range, overhang moment loads, running torque, and static torque. This latter term pertains to protection of the engine or transmission, which drives the accessory, from damage to be expected from malfunction of the accessory. This protection is usually supplied by providing a shear section in the accessory drive shaft designed to fail before exceeding the static torque limit of the engine or transmission driving component. Note that when evaluating the strength of the mechanical shear section, material allowables quoted in materials handbooks should not be used since these are minimum strength values. Shear sections should consider maximum strength values to be expected which are on the order of 130 percent of the minimum strength values. Also, it should be verified that design data for shear sections are dimensioned to limit the maximum diameter as well as the minimum diameter. Installation of starter-generators may also require verification that horsepower extraction limits are not exceeded. Special flightcrew instructions in the flight manual to monitor generator load or to disconnect electrically loaded items to protect accessory or engine-transmission pad limits should be avoided. Environmental qualification requires consideration or protection against adverse effects of heat, sand or dust, humidity and rain, salt-laden atmosphere, and extremes of cold weather. Accessories such as generators, pumps, etc., are subjected to many of these aspects during the individual qualification tests; however, satisfactory overall integrated system performance under these adverse conditions should be verified. Cold weather testing should include verification that lubricating oils and greases function properly and that engine starting procedures are safe and do not impose excessive loads on accessories, engines, or drive system components. Powerplant engineers should coordinate compliance efforts in this area with the system engineer's investigations of compliance with §§ 29.1301 and 29.1309. Full-scale rotorcraft operations in cold weather should be required. Performance tests are required at the minimum temperature to be certified. Propulsion systems may usually be evaluated at this time. Cold soak or overnight exposure to cold weather is appropriate followed by starting and pretakeoff procedures in accordance with the flight manual. Attention should be given to the practicality of important mandatory inspection procedures as affected by cold weather.

(iii) Accessibility. Accessibility for maintenance should be reviewed. Typically, some maintenance activities must involve disassembly or removal of adjacent components. This should be avoided if repetitive activity can jeopardize the performance of critical or safety-related equipment. Verify that easy access exists to items such as oil system sight gauges or dip sticks, filler ports and drain valves for engines, auxiliary propulsion units, transmissions, fuel tanks and filters, etc.

(iv) Electrical (Grounding). Electrical interconnections to prevent difference of potential should be provided in the form of grounding straps or wires sized to carry the currents to be expected. Verify that the attachments for these grounding devices are not compromised by paint or zinc chromate which will tend to electrically insulate the engine or component. Note that engine mount structure should not be accepted as a grounding device since electrical current will cause corrosion at attachment points.

(v) Thermal Expansion. Axial and radial expansion of turbine engines is usually not a problem unless redundant mount arrangements are used. Special expansion provisions are usually required if engine components other than mounting points are attached to bulkheads, firewalls, other engines, or drive system components. Engine output shaft axial or bending loads due to thermal expansion and to deflection of supports under ground or flight loads should be checked. Other components of concern are compressor inlet flanges, exhaust ducts, and rigid fluid or air lines between aircraft structure and the engine. The engine installation data will provide limit loads to be considered for parts of the engine which normally are attached to airframe components.

c. Section 29.901(c):

(1) Explanation. Paragraph (c) requires, with notable exceptions, a detailed failure modes and effects analysis (FMEA) of the various powerplant systems and components to establish that anticipated failures will not jeopardize the safe operation of the rotorcraft. Alternative methods such as top-down analysis may also be used.

Exceptions include engine rotor discs and structural elements for which the probability of failure can be shown to be “extremely remote.” Items in this latter case would include all components of the rotor drive system evaluated under § 29.571 provided that the reliability of any item or system exempted under § 29.901(c)(1) is not jeopardized by the failure of other systems/components which themselves may be less reliable than “extremely remote.” Items of consideration here would include, but not be limited to, powerplant cooling systems, probable maintenance errors, deterioration/failure of seals and other time/temperature/weather sensitive nonmetallics, high energy fragment impact damage of nearby dynamic components, etc. Some items in these categories are addressed by specific rules in this subpart which override consideration under § 29.901(c). For example, § 29.927 sets forth specific tests to demonstrate acceptable safety levels in event of overtorque, overspeed, and transmission lubrication system failures. Further consideration of failures in these areas (under § 29.901(c)(1)) probably would be inappropriate. It would not, however, be appropriate to assume that an engine certified under Part 33, an auxiliary power unit qualified under TSO C-77, or other components qualified under various TSO’s or military specifications would not be subject to failure. As a general rule, any component or system whose failure is “probable” and the failure, in conjunction with probable combinations of failures, significantly degrades safe operation and/or impairs the capability of the crew to operate the rotorcraft safely constitutes an apparent noncompliance unless it is compensated for by alternate components, systems, or if appropriate, special operating procedures which essentially restore a safe level of operation of the rotorcraft. Normally, safe “continued” flight is intended; however, for the special case of the single-engine rotorcraft, safe entry into autorotation after engine failure is an acceptable means of compliance provided that other coincidental or associated failures or malfunctions do not jeopardize this maneuver.

(2) Procedures.

(i) The general techniques of AC 25.1309-1, System Design Analysis, present an acceptable means of evaluating the powerplant systems/components for compliance. However, the quantitative assessments of the probability classifications in AC 25.1309-1 have not been universally adopted for powerplant systems and components. Other procedural techniques in AC 25.1309-1 may be impractical for powerplant systems. This does not preclude using a similar but simplified methodology in conjunction with conservative engineering judgment to arrive at a determination of compliance or identification of noncompliance aspects, using the following as a guide (extracted from AC 25.1309-1). Develop a matrix of all applicable powerplant components/systems which includes:

(A) Possible modes of failure, including malfunctions and damage from external sources.

(B) The probability of multiple failures and undetected failures.

(C) The resulting effects of the rotorcraft and occupants, considering the stage of flight and operating conditions, and

(D) The crew warning cues, corrective action required, and the capability of detecting faults.

(ii) Prepare an item-by-item, system-by-system FMEA. The analysis to identify failure conditions should be qualitative. An assessment of the probability of a failure condition can be qualitative or quantitative. An analysis may range from a simple report which interprets test results or presents a comparison between two similar systems to a fault/failure analysis which may (or may not) include numerical probability data. An analysis may make use of previous service experience from comparable installations in other aircraft.

(iii) Powerplant engineers normally find that believable statistical failure data on powerplant components are not readily available. Therefore, the simpler form of analysis involving assumption of failure with either benign results or dependence on alternate or redundant systems/components becomes the most feasible method of finding compliance. Repetitive inspections and preflight checks are a significant part of this finding, particularly if the backup system/component is used or checked routinely in the operation of the rotorcraft.

d. Section 29.901(d):

(1) Explanation. This paragraph provides a generalized basis for requiring compliance with any rules in this Part applicable to safe installation and operation of auxiliary power units (APU's). The wording of the rule is generalized to permit (and require) a detailed review of this Part to identify any existing rule related to this type of equipment. Generally, any rule related to engines and their installation, support systems, and fire protection should be considered to be applicable to APU's. This review may result in a designation of "nonapplicable" to certain engine-related rules if limitations such as "ground-use-only" are applied or if the APU serves only nonessential services. Any questionable aspects or interpretation/policy involved in establishing the applicable rules should be coordinated with the FAA Aircraft Certification Office. Notwithstanding the generalization discussed above, a number of specific rules in subparts E and F include reference to APU's in their applicability. The presence of these references should not be interpreted as excluding applicability of other appropriate rules as discussed above. In addition, the APU itself must be shown to be safe and reliable. Normally, this aspect is satisfied by showing that the APU model is included in the qualified parts list of TSO-C77a. This TSO also requires establishment (by the APU manufacturer) of limitations and installation data peculiar to the model APU. A showing of compliance with these data for the APU installed in the rotorcraft will be expected.

(2) Procedures.

(i) Verify that the Model APU is listed as qualified to TSO-C77(a) or other suitable specifications. Note that TSO qualification is not regulatory but simply defines an acceptable base qualification standard. Other standards may be acceptable or deviations from the TSO may be acceptable if evaluated and found not pertinent to the planned installation.

(ii) Review the installation data provided for the APU and determine that the installation is in compliance. Exceptions may be taken as discussed above. Note that the TSO provides different qualification standards for “essential” and “nonessential” service APU’s. However, it does not distinguish between “flight-use” and “ground-use-only” APU’s. Some deviations to the TSO may be authorized based on this aspect; i.e., operation during negative “g” conditions.

(iii) Review Part 29, especially subparts E and F for all rules related to engines, engine support/service systems, intakes, exhausts, instrumentation, fire protection, pneumatic systems, etc., for applicability to installation and operation of the APU. Develop and accomplish a compliance program for the rules identified by this review following policy and procedures used for engines with exceptions which may be justified as discussed above.

(iv) For reference, the following rules specifically refer to APU’s. Some comments regarding compliance are offered.

(A) Section 29.1041, Cooling. APU installation data should define limits to be substantiated.

(B) Section 29.1091, Air Induction. Note the requirements of paragraph (f).

(C) Section 29.1103, Induction System Ducts. Note the special requirements of paragraphs (a), (e), and (f).

(D) Section 29.1121, Exhaust Systems.

(E) Section 29.1142, Controls.

(F) Section 29.1181, Designated Fire Zones.

(G) Section 29.1191, Firewalls. Firewall construction should be provided to completely separate the APU from other parts of the rotorcraft.

(H) Section 29.1195, Fire Extinguishers. Note that only one adequate discharge is required.

(I) Section 29.1203, Fire Detector Systems. Detectors are required for each fire zone which would include APU installations.

(J) Section 29.1305, Powerplant Instruments. TSO-C77(a) specifies provisions for measuring gas temperature, rotor RPM, and any other parameter necessary for safe operation of the APU.

(K) Section 29.1337, Powerplant Instruments.

(v) Additional comments. APU fuel sources which tap into engine fuel systems should be carefully designed and arranged to minimize the probability that an APU fuel line failure will jeopardize continued normal engine operation. If the APU provides essential services, it should be provided with an independent fuel system. Also, engine fuel systems which operate at negative pressures should not be tapped for APU fuel source since air leaks back through the APU fuel control or small leaks in the APU fuel system likely will fail the engine.

AC 29.901A. § 29.901 (Amendment 29-26) INSTALLATION.

a. Explanation. Amendment 29-26 changes § 29.901(b)(2) to require a satisfactory determination that rotorcraft can operate safely throughout adverse environmental conditions such as high altitude and temperature extremes. This amendment was needed to provide consistent application of environmental qualification aspects. This amendment also added a new paragraph § 29.901(b)(6) to require design precautions to minimize the potential for incorrect assembly of components and equipment essential to safe operation.

b. Procedures. All of the policy material pertaining to this section remains in effect with the addition of design precautions. Design precautions should be taken to minimize the possibility of improper assembly of components essential to the safe operation of the rotorcraft. Fluid lines, electrical connectors, control linkages, etc., should be designed so that they cannot be incorrectly assembled. This can be achieved by incorporating different sizes, lengths, and types of connectors, wires, fluid lines, and mounting methods.

AC 29.901B. § 29.901 (Amendment 29-36) INSTALLATION.

a. Explanation. Prior to Amendment 29-36, paragraph (c) exempted engine rotor disc failures (engine rotorburst) from consideration as a failure that could jeopardize the safe operation of the rotorcraft. Amendment 29-36 removes this exclusion. Therefore, engine rotor disc failures should be considered as a failure that would jeopardize the safe operation of the rotorcraft.

b. Procedures. The method of compliance for this section is unchanged.

AC 29.903. § 29.903 (Amendment 29-12) ENGINES.

a. Explanation. While paragraph (a) of this section requires engines to be type certificated under Part 33 of this chapter, engines certificated under other approved certification rules (CAR Part 13 and § 21.29 for imported engines) are also eligible. The fact that a component, system, or arrangement for which Part 29 standards exist is approved as a part of a certificated engine should not, except when specifically stated in Part 29, relieve an applicant of the necessity for compliance with Part 29. Even if the component, system, or arrangement supplied as a part of a certificated engine does meet the Part 29 standard, the possibility that subsequent changes to these components, systems, or arrangements by the engine manufacturer could negate compliance with Part 29 must be considered. For example, an engine may initially be equipped by the engine manufacturer with an oil tank filler cap that meets the Category A requirements of § 29.1013(c)(2) but is subsequently changed to a simpler and less expensive cap complying with § 33.71(c)(4). Continued monitoring of the engine configuration by the rotorcraft certification team would be needed to preclude an occurrence of noncompliance.

b. Procedures.

(1) Category A; Engine Isolation. This rule is one of the most significant safety rules in Subpart E of Part 29. Compliance involves a very extensive and rigorous evaluation not only of essentially all systems of the rotorcraft, but of the controls, both flight and powerplant, instruments, cockpit arrangement, cockpit switches, and operating procedures. A complete failure modes and effects analysis is involved. Section 29.903(b)(1) should be rigorously applied to rotorcraft engine control arrangements which utilize governors responding to main rotor speed to modulate power rather than power levers preset to produce equal or less than limit power. Section 29.903(b)(2) precludes "immediate action by any crewmember for continued safe operation." This should be interpreted as requiring all powerplant systems to operate safely and continuously without crew attention (except to maintain flight using primary flight controls) in event of an engine failure from any cause, including fire. The collective is considered a primary flight control and not a powerplant control even though collective movement affects engine operation. No adjustment to powerplant controls or configuration can be allowed for certification purposes for performance credit or for safety. The time increment associated with "immediate" action may vary among different designs; however, it must not be less than that required to established engine-out flight profiles and climb rates associated with Category A performance. During critical takeoff flight regimes, flight translation to at least published takeoff safety speed is needed before crew attention can be mandated to modulate powerplant controls or change aircraft configuration (i.e., landing gear, power lever or rotorspeed governor setting, etc.) to achieve published flight performance. This does not mean crew action is prohibited--only that no credit for crew action can be allowed for any resulting improved performance in the performance section of the flight manual.

(2) Category A; Control of Engine Rotation.

(i) Means for stopping any engine in flight is to be considered unless it is shown that after critical failure of the engine, or components/accessories driven by the engine (not including rotor drive system components), no hazard results from rotation during the coast-down period. If continued rotation occurs, no hazard should result due to rotation during the period that the rotation is expected to continue. (Consider unbalanced rotors, bearing failures, accessory failures, lack of lubrication to other engine rotors, etc.) Note that after emergency engine shutdown, coast-down and continued rotation speed can be influenced by ram air flow into the compressor and, for multiengine rotorcraft, drag through the freewheeling unit.

(ii) A requirement exists for Category A rotorcraft to incorporate a means for restarting any engine individually in flight. Compliance is usually obtained during official flight tests and/or applicant tests in accordance with an approved test plan by requiring actual engine air-start demonstrations to define an acceptable restart envelope. These air-starts should be conducted at various altitudes, ambient temperatures, and fuel temperatures using the fuel type most critical, unless the applicant can show that this parameter is not pertinent. Other concerns involve the pilot station arrangement for flight controls and engine starting controls; i.e., verify that the engine start can be accomplished without jeopardizing continued safe operation of the rotorcraft, considering the pilot workload for the preexisting one-engine-inoperative situation, the location of the restart system controls, availability of a second pilot, etc. Also, verify that the emergency/malfunction instruction sections of the RFM present a detailed definition of the approved restart envelope and detailed instructions for the restart, including eligible ambient atmospheric conditions, prestart arrangement of fuel, electrical and pneumatic systems (as applicable), delay time between start attempts (to allow for waste fuel drainage), starter duty cycle (if different from ground start duty cycle), and prestart situation analysis (i.e., Should a restart be attempted in view of the cause for initial shutdown? Is inlet system ice ingestion a possibility? Is reignition of fuel in the engine nacelle a possibility? Is sufficient restart time available? Is power available and is altitude sufficient to maintain terrain clearance?). Although restart capability from an all-engines-out flight condition is not required, special instructions for restarting from this situation should also be included commensurate with the system capability to accomplish the starts.

(3) Although restart capability is required for only Category A rotorcraft, the applicant should be encouraged to provide air start instructions in accordance with the above criteria for both single and multiengine Category B rotorcraft, including all-engine-out instructions if reasonable and practicable.

c. Turbine Engine Installation.

(1) Explanation. The certification of turbine engines and particularly the qualification of turbine rotors assume that the limitations established during these certifications will be accurately and rigorously observed during ground and flight operations in an aircraft. This paragraph is intended to promote this concept.

(2) Procedures. Primary engine limitations in the form of time, gas temperature, torque, and rotational speed and their corresponding allowable transient values are defined in the approved engine installation manual. The rotorcraft manufacturer must provide reliable, accurate means to assure that these limitations are not exceeded. These means may be in the form of automatic limiters or by crew monitoring of appropriately marked instruments. The FAA/AUTHORITY powerplant certification engineer and the rotorcraft manufacturer's staff should verify these aspects by:

(i) Evaluating all applicable instrument, indicator, or warning devices, including transmitters, and limiting devices, if any, for system tolerances.

(ii) Closely reviewing the component qualification reports of items in c(2)(i) above to verify that these devices are properly qualified and that any deviations are acceptable.

(iii) Assuring that maintenance data are provided for functional checks and calibration of instruments and devices which are used to monitor or protect critical turbine rotor limitations. Preflight checks for automatic limiter devices may be appropriate.

(iv) Verifying that instrument markings are clear and relatively simple, that corresponding flight manual instructions and descriptions are straightforward and complete, and that instruments are located and orientated to minimize the probability of reading error.

AC 29.903A. §29.903 (Amendment 29-26) ENGINES.

a. Explanation. Amendment 29-26 adds § 29.903(a) that requires reciprocating engines used in rotorcraft to be certified in accordance with the rotorcraft engine testing requirements in § 33.49(d). This change is incorporated to ensure that certification requirements are not overlooked when reciprocating engines are installed in rotorcraft to be certified under Part 29 requirements. Section 29.903(b)(2) was revised to identify and clarify crew action; i.e., normal pilot action allowable with primary flight controls, in determining if adequate powerplant systems isolation is provided. This change 0eliminates any possible confusion that may exist regarding the acceptability of modifying optimum flight control manipulation to protect engine parameters. Section 29.903(c)(3) was added and requires engine restart capability to be available throughout the flight envelope appropriate to the rotorcraft. This will avoid the concept that an in-flight engine restart envelope constitutes acceptable compliance with this rule.

b. Procedures.

(1) Engine type certification. All engines installed in rotorcraft should have a type certificate. The specific certification requirements for installation of reciprocating engines in rotorcraft are found in Part 33. Engines certificated under other approved certification rules (CAR Part 13 and FAR § 21.29, for imported engines) are also eligible. If a component, system, or arrangement is certified under Part 33 or other requirement, the applicant is not relieved of the necessity to comply with the requirements of Part 29. If the component, system, or arrangement, supplied as a part of a certificated engine, meets the Part 33 and Part 29 requirements, subsequent changes to these components, systems, or arrangements could negate compliance with Part 29. For example, an engine may initially be equipped by the engine manufacturer with an oil tank filler cap that complies with the Category A requirements of § 29.1013(c)(2) but is subsequently changed to a simpler and less expensive cap that complies with § 33.71(c)(4). The airframe manufacturer should ensure that the requirements of § 29.1013(c)(2) are maintained.

(2) Category A: control of engine rotation. Section 29.903(c)(3) requires an engine restart capability which is appropriate to the rotorcraft. The minimum envelope for the restart capability should be equal to or better than the rotorcraft takeoff/landing maximum altitude and temperature limits. Compliance is usually shown by conducting actual in-flight restarts during flight tests and/or other tests in accordance with an approved test plan. Restarts should be conducted at various altitudes, ambient temperatures, and fuel temperatures using the fuel type most critical, unless the applicant can show that this parameter is not pertinent. Other concerns involve the pilot station arrangement for flight controls and engine starting controls. It should be verified that the engine start can be accomplished without jeopardizing continued safe operation of the rotorcraft. Pilot workload for a preexisting one-engine-inoperative situation, the location of the restart system controls, and the availability of a second pilot should be considered. The emergency/malfunction instruction sections of the rotorcraft flight manual (RFM) should present a detailed definition of the approved restart envelope and detailed instructions for the restart. Eligible ambient atmospheric conditions, prestart requirements (to allow for waste fuel drainage), starter duty cycle (if different from the ground start duty cycle), and prestart situation analysis should be included. The prestart situation analysis should consider the following questions:

– Should a restart be attempted in view of the cause for initial shutdown?
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– Is inlet system ice ingestion a possibility?
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– Is reignition of fuel in the engine nacelle a possibility?
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– Is sufficient restart time available?
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– Is power available?
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– Is altitude sufficient to maintain terrain clearance?

Although restart capability from an all-engines-out flight condition is not required, special instructions for restarting from this situation should also be included commensurate with the system capability to accomplish the starts.

AC 29.903B. <u>§ 29.903 (Amendment 29-31) ENGINES</u>.

a. <u>Explanation</u>. Amendment 29-31 clarified the requirements for control of engine rotation and in-flight restart of engines. Section 29.903(c)(1) was changed by adding the word “or” at the end of the paragraph, which provided an option on how to protect the engine stopping system from fire.

b. <u>Procedures</u>. All of the policy material pertaining to this section remains in effect. Additionally, the new § 29.903(e) requires that any engine should have a restart capability that has been demonstrated throughout a flight envelope to be certificated for the rotorcraft. The minimum restart envelope for Category A rotorcraft is discussed in paragraph AC 29.903A. The restart capability can consider windmilling of the engine as part of this restart capability; however, most rotorcraft airspeeds and the locations of the engines do not support engine windmilling up to start speeds. Only electrical power requirements were considered for restarting; however, other factors that may affect this capability are permitted to be considered. Engine restart capability following an in-flight shutdown of all engines is the primary requirement, and the means of providing this capability is left to the applicant.

AC 29.903C. <u>§ 29.903 (Amendment 29-36) TURBINE ENGINE INSTALLATION</u>.

a. <u>Explanation</u>. Amendment 29-36 revises § 29.903(d) to require that design precautions should be taken to minimize hazards to the rotorcraft in the event of an engine failure. This advisory material sets forth a method of compliance with the requirements of §§ 29.901, 29.903(b)(1), and 29.903(d)(1) of the Federal Aviation Regulations (FAR) pertaining to design precautions taken to minimize the hazards to rotorcraft in the event of uncontained engine rotor (compressor and turbine) failure. It is for guidance and to provide a method of compliance that has been found acceptable. As with all AC material, it is not mandatory and does not constitute a regulation.

b. <u>Procedures.</u> Although turbine engine manufacturers are making efforts to reduce the probability of uncontained rotor failures, service experience shows that such failures continue to occur. Failures have resulted in high velocity fragment penetration of fuel tanks, adjacent structures, fuselage, system components and other engines of the rotorcraft. Since it is unlikely that uncontained rotor failures can be completely eliminated, rotorcraft design precautions should be taken to minimize the hazard from such events. These design precautions should recognize rotorcraft design features that may differ significantly from that of an airplane, particularly regarding an engine location and its proximity to another engine or to other systems and components.

(1) Uncontained gas turbine engine rotor failure statistics for rotorcraft are presented in the Society of Automotive Engineers (SAE) Report No.'s AIR 4003 (period 1976-83) and AIR 4770 (period 1984-89).

(2) The statistics in the SAE studies indicate the existence of some failure modes not readily apparent or predictable by failure analysis methods. Because of the variety of uncontained rotor failures, it is difficult to analyze all possible failure modes and to provide protection to all areas. However, design considerations outlined in this AC provide guidelines for achieving the desired objective of minimizing the hazard to rotorcraft from uncontained rotor failures. These guidelines, therefore, assume a rotor failure will occur and that analysis of the effects or evaluation of this failure is necessary. These guidelines are based on service experience and tests but are not necessarily the only means available to the designer.

c. Definitions.

(1) Minimize. Reduce to the least possible amount by means that can be shown to be both technically feasible and economically justifiable.

(2) Separation. Positioning of redundant critical structure, systems, or system components within the impact area such that the distance between the components minimizes the potential impact hazard. Redundant critical components should be separated within the spread angles of a rotor by a distance at least equal to either a ½ unbladed disk (hub, impeller) sector, or a 1/3 bladed disk (hub, impeller) sector with 1/3 blade height, with each rotating about its center of gravity (CG), whichever is greater (see figure AC 29.903C-6).

(3) Isolation. A means to limit system damage so as to maintain partial or full system function after the system has been damaged by fragments. Limiting the loss of hydraulic fluid by the use of check valves to retain the capability to operate flight controls is an example of "isolation." System damage is confined allowing the retention of critical system functions.

(4) Rotor.

(i) Rotor means the rotating components of the engine and APU that analysis, test results, and/or experience has shown can be released during uncontained failure with sufficient energy to hazard the rotorcraft.

(ii) The engine or APU manufacturer should define those components that constitute the rotor for each engine and APU type design. Typical rotors have included, as a minimum, disks, hubs, drums, seals, impellers, and spacers.

(5) Uncontained Engine or APU Failure (or Rotorburst). For the purposes of rotorcraft evaluations in accordance with this AC, uncontained failure of a turbine engine is any failure which results in the escape of rotor fragments from the engine or

APU that could create a hazard to the rotorcraft. Rotor failures of concern are those in which released fragments have sufficient energy to create a hazard to the rotorcraft. Uncontained failures of APU's which are "ground operable only" are not considered hazardous to the rotorcraft.

(6) Critical Component (System). A critical component is any component or system whose failure or malfunction would contribute to or cause a failure condition that would prevent the continued safe flight and landing of the rotorcraft. These components (systems) should be considered on an individual basis and in relation to other components (systems) that could be degraded or rendered inoperative by the same fragment or by other fragments during any uncontained failure event.

(7) Fragment Spread Angle. The fragment spread angle is the angle measured, fore and aft, from the center of the plane of rotation of the disk (hub, impeller) or other rotor component initiating at the engine or APU shaft centerline or axis of rotation (see figure AC 29.903C-1). The width of the fragment should be considered in defining the path of the fragment envelope's maximum dimension.

(8) Ignition Source. Any component that could precipitate a fire or explosion. This includes existing ignition sources and potential ignition sources due to damage or fault from an uncontained rotor failure. Potential ignition sources include hot fragments, damage or faults that produce sparking, arcing, or overheating above the auto-ignition temperature of the fuel. Existing ignition sources include items such as unprotected engine or APU surfaces with temperature greater than the auto-ignition temperature of the fuel or any other flammable fluid.

d. Safety Assessment.

(1) Procedure. Assess the potential hazard to the rotorcraft using the following procedure:

(i) Minimizing Rotorburst Hazard. The rotorburst hazard should be reduced to the lowest level that can be shown to be both technically feasible and economically justifiable. The extent of minimization that is possible will vary from new or amended certification projects and from design to design. Thus the effort to minimize must be determined uniquely for each certification project. Design precautions and techniques such as location, separation, isolation, redundancy, shielding, containment and/or other appropriate considerations should be employed, documented, agreed to by the certifying authority, and placed in the type data file. A discussion of these methods and techniques follows.

(ii) Geometric Layout and Safety Analysis. The applicant should prepare a preliminary geometric layout and safety analysis for a minimum rotorburst hazard configuration determination early in the design process and present the results to the certification authority no later than when the initial design is complete. Early contact and coordination with the certifying authority will minimize the need for design modification later in the certification process. The hazard analysis should follow the guidelines indicated in paragraphs AC 29.901c(2) and AC 29.903Cd(6). Geometric layouts and analysis should be used to evaluate and identify engine rotorburst hazards to critical systems, powerplants, and structural components from uncontained rotor fragments, and to determine any actions which may be necessary to further minimize the hazard. Calculated geometric risk quantities may be used in accordance with paragraph d(4) following, to define the rotorcraft configuration with the minimum physical rotorburst hazard.

(2) Engine and APU Failure Model. The safety analysis should be made using the following engine and APU failure model, unless for the particular engine/APU type concerned, relevant service experience, design data, test results or other evidence justify the use of a different model. In particular, a suitable failure model may be provided by the engine/APU manufacturer. This may show that one or more of the considerations below do not need to be addressed.

(i) Single One-Third Disc Fragment. It should be assumed that the one-third disc fragment has the maximum dimension corresponding to one-third of the disc with one-third blade height and a fragment spread angle of ±3°. Where energy considerations are relevant, the mass should be assumed to be one-third of the bladed disc mass and its energy--the translational energy (i.e., neglecting rotational energy) of the sector (see figure AC 29.903C-2).

(ii) Intermediate Fragments. It should be assumed that the intermediate fragment has a maximum dimension corresponding to one third or the disc radius with one-third blade height and a fragment spread angle of ±5°. Where energy considerations are relevant, the mass should be assumed to be 1/30th of the bladed disc mass and its energy--the translational energy (neglecting rotational energy) of the piece traveling at rim speed (see figure AC 29.903C-3).

(iii) Alternative Engine Failure Model. For the purpose of the analysis, as an alternative to the engine failure model of paragraphs d(2)(i) and d(2)(ii) above, the use of a single one-third piece of disc having a fragment spread angle of ±5° would be acceptable, provided that the objectives of the analysis are satisfied.

(iv) Small Fragments. It should be assumed that small fragments have a maximum dimension corresponding to the tip half of the blade airfoil and a fragment spread angle of ±15°. Where energy considerations are relevant, the mass should be assumed to be corresponding to the above fragment dimensions and the energy is the translational energy (neglecting rotational energy) of the fragment traveling at the speed of its CG location. The effects of multiple small fragments should be considered during this assessment.

(v) Critical Engine Speed. Where energy considerations are relevant, the uncontained rotor event should be assumed to occur at the engine shaft speed for the maximum rating appropriate to the flight phase (exclusive of OEI ratings), unless the most probable mode of failure would be expected to result in the engine rotor reaching a red line speed or a design burst speed. For APU's, use the maximum rating appropriate to the flight phase or the speed resulting from a failure of any one of the normal engine control systems.

(vi) APU Failure Model. Service experience has shown that some APU rotor failures produced fragments having significant energy to have been expelled through the APU tailpipe. For the analysis, the applicable APU service history and test results should be considered in addition to the failure model as discussed in paragraph d(2) above for certification of APU installations near critical items. In addition, the APU installer needs to address the rotorcraft hazard associated with APU debris exiting the tailpipe. Applicable service history or test results provided by the APU manufacturer may be used to define the tailpipe debris size, mass, and energy. The uncontained APU rotor failure model is dependent upon the design/analysis, test results and service experience.

(A) For APU's in which rotor integrity and blade containment have been demonstrated in accordance with TSO-C77a/JAR APU, i.e., without specific containment testing, paragraphs d(2)(i), d(2)(ii), and d(2)(iv) or paragraph d(2)(iii) and d(2)(iv) apply. If shielding of critical airframe components is proposed, the energy level that should be considered is that of the tri-hub failure released at the critical speed as defined in paragraph d(2)(v). The shield and airframe mounting point(s) should be shown to be effective at containing both primary and secondary debris at angles specified by the failure model.

(B) For APU rotor stages qualified as contained in accordance with the TSO, an objective review of the APU location should be made to ensure the hazard is minimized in the event of an uncontained APU rotor failure. Historical data shows that in-service uncontained failures have occurred on APU rotor stages qualified as contained per the TSO. These failure modes have included bi-hub and overspeed failure resulting in some fragments missing the containment ring. In order to address these hazards, the installer should use the small fragment failure model, or substantiated in-service data supplied by the APU manufacturer. Analytical substantiation for the shielding system if proposed is acceptable for showing compliance.

(3) Engine/APU Rotorburst Data. The engine or APU manufacturer should provide the required engine data to accomplish the evaluation and analysis necessary to minimize the rotorburst hazard such as:

(i) Engine failure model (range of fragment sizes, spread angles and energy).

(ii) Engine rotorburst probability assessment.

(iii) List of components constituting the rotors.

(4) Fragment Impact Risks. FAA/AUTHORITY research and development studies have shown that, for rotorcraft conventional configurations (one main rotor and one tail rotor), the main and tail rotorblades have minimal risks from a rotorburst, and thus, they require no special protection. However, unique main and tail rotor blade configurations should be carefully reviewed. Certain zones of the tail rotor drive shaft and other critical parts which may be necessary for continued safe flight and landing may not have natural, minimal risk from uncontained rotor fragments.

(5) Engine Service History/Design.

(i) For the purpose of a gross assessment of the vulnerability of the rotorcraft to an uncontained rotorburst, it must be taken that an uncontained engine rotor failure (burst) will occur. However, in determining the overall risk to the rotorcraft, engine service history and engine design features should be included in showing compliance with § 29.903 to minimize the hazard from uncontained rotor failures. This is extremely important since the engine design and/or the service history may provide valuable information in assessing the potential for a rotorburst occurring and this should be considered in the overall safety analysis.

(ii) Information contained in the recent SAE studies should be considered in this evaluation (see paragraph b(1) above).

(6) Certification Data File. A report, including all geometric layouts, that details all the aspects of minimizing the engine rotorburst hazards to the rotorcraft should be prepared by the applicant and submitted to the certification authority. Items which should be included in this report are the identification of all hazardous failures that could result from engine rotor failure strikes and their consequences (i.e., an FMEA or equivalent analysis) and the design precautions and features taken to minimize the identified hazards that could result from rotor failure fragment strikes. Thus an analysis that lists all the critical components; quantifies and ranks their associated rotorburst hazard; and clearly shows the minimization of that quantified, ranked hazard to the "maximum practicable extent" should be generated and agreed upon during certification. Critical components should all be identified and their rotorburst hazard quantified, ranked, and minimized where necessary. Design features in which the design precautions of this guidance material are not accomplished should be identified along with the alternate means used to minimize the hazard. To adequately address minimizing the hazards, all rotorcraft design disciplines should be involved in the applicant's compliance efforts and report preparation.

e. Design Considerations. Practical design precautions should be used to minimize the damage that can be caused by uncontained engine and APU rotor debris. The following design considerations are recommended:

(1) Consider the location of the engine and APU rotors relative to critical components, or areas of the rotorcraft such as:

(i) Opposite Engine - Protection of the opposite engine from damage from 1/3 disc rotor fragments may not be feasible. Protection of the opposite engine from other fragments may be provided by locating critical components, such as engine accessories essential for proper engine operation (e.g. high pressure fuel lines, engine controls and wiring, etc.), in areas where inherent shielding is provided by the fuselage, engine, or other structure.

(ii) Engine Controls - Controls for the remaining engine(s) that pass through the uncontained engine failure zone should be separated/protected to the maximum extent practicable.

(iii) Primary Structure of the Fuselage.

(iv) Flight Crew - The flight crew is considered a critical component.

(v) Fuel system components, piping and tanks, including fuel tank access panels (NOTE: Spilled fuel into the engine or APU compartments, on engine cases or on other critical components or areas could create a fire hazard.)

(vi) Critical control systems, such as primary and secondary flight controls, electrical power cables, systems and wiring, hydraulic systems, engines control systems, flammable fluid shut-off valves, and the associated actuation wiring or cables.

(vii) Engine and APU fire extinguisher systems including electrical wiring and fire extinguishing agent plumbing to engine and APU compartments.

(viii) Instrumentation necessary for continued safe flight and landing.

(ix) Transmission and rotor drive shafts.

(2) Location of Critical Systems and Components. The following design practices have been used to minimize hazards to critical components:

(i) Locate, if possible, critical components or systems outside the likely debris impact areas.

(ii) Duplicate and separate critical components or systems if located in debris impact areas or provide suitable protection.

(iii) Protection of critical systems and components can be provided by using airframe structure where shown to be suitable.

(iv) Locate fluid shutoffs so that flammable fluids can be isolated in the event of damage to the system. Design and locate the shut-off actuation means in protected areas or outside debris impact areas.

(v) Minimize the flammable fluid spillage which could contact an ignition source.

(vi) For airframe structural elements, provide redundant designs or crack stoppers to limit the subsequent tearing which could be caused by uncontained rotor fragments.

(vii) Consider the likely damage caused by multiple fragments.

(viii) Fuel tanks should not be located in impact areas. However, if necessitated by the basic configuration requirements of the rotorcraft type to locate fuel tanks in impact areas, then the engine rotorburst hazard should be minimized by use of design features such as minimization of hazardous fuel spillage (that could contact an ignition source by drainage or migration); by drainage of leaked fuel quickly and safely into the airstream; by proper ventilation of potential spillage areas; by use of shielding; by use of explosion suppression devices (i.e., explosion resistant foam or inert gases); and by minimization of potential fuel ignition sources or by other methods to reduce the hazard.

(ix) The rotor integrity or containment capability demonstrated during APU evaluation to TSO-C77a, or JAR-APU should be considered for installation certification.

(x) The flight data recorder, cockpit voice recorder, and emergency locator transmitter, if required, should be located outside the impact zone when practical.

(xi) Items such as human factors, pilot reaction time, and correct critical system status indication in the pilot compartment after an uncontained engine failure has occurred should be considered in design to permit continued safe flight and landing.

(3) <u>Rotorcraft Modifications</u>. Modifications made to rotorcraft certified to this rule should be assessed with the considerations of this AC. These modifications include but are not limited to re-engining installations (including conversion from reciprocating to turbine powered), APU installations, fuselage stretch, and auxiliary fuel tank installations. Auxiliary fuel tank(s) should be located as much as practical so as to minimize the risk that this tank(s) will be hit by rotor failure fragments. The need to remain within the approved CG limits of the aircraft will of necessity limit the degree to which the risk may be minimized.

f. <u>Protective Measures</u>. The following list is provided for consideration as some measures which may be used to minimize effects of a rotorburst:

(1) Powerplant Containment.

(i) Engine Rotor Fragment Containment. It should be clearly understood that containment of rotor fragments is not a requirement. However, it is one of many options which may be used to minimize the hazards of an engine rotorburst. Containment structures (either around the engine, or APU, or on the rotorcraft) that have been demonstrated to provide containment should be accepted as minimizing the hazard defined by the rotor failure model for that particular rotor component. Contained rotor in-service failures may be used to augment any design or test data. Containment material stretch and geometric deformation should be considered in conjunction with fragment energies and trajectories in defining the hazards to adjacent critical components such as structures, system components, fluid lines, and control systems. Data obtained during containment system testing along with analytical data and service experience should be used for this evaluation.

(ii) APU Containment. Rotor integrity or containment capability demonstrated during APU TSO evaluation should be considered for installation certification. If rotor containment option was shown by analysis or rig test an objective review of the APU location should be made to ensure the hazard is minimized in the event of an uncontained APU rotor failure.

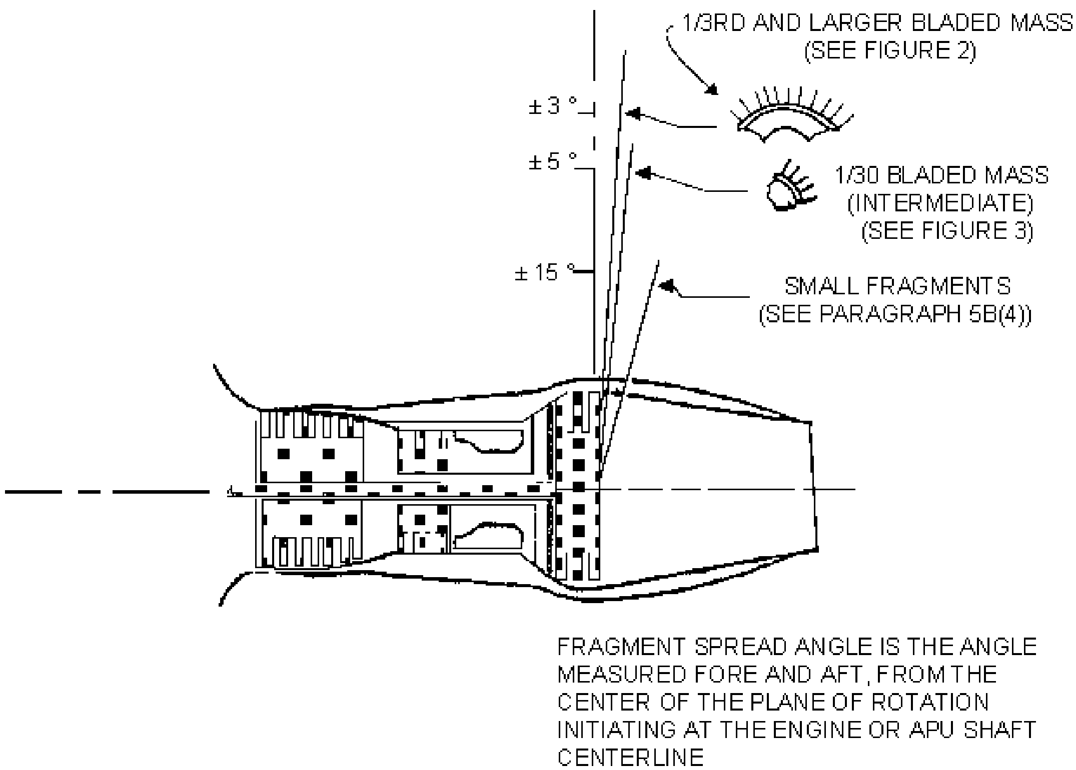
(2) Shields and Deflectors. When shields, deflection devices, or intervening rotorcraft structure are used to protect critical systems or components, the adequacy of the protection should be shown by testing or analysis supported by test data, using the impact area, fragment mass, and fragment energies based on the definitions stated herein. Analytical methods used to compute protective armor or shielding thicknesses and energy absorption requirements should reflect established methods, acceptable to the certifying authority, that are supported by adequate test evidence. Protective armor, shielding, or deflectors that stop, slow down, or redirect uncontained fragments redistribute absorbed energy into the airframe. The resulting loads are significant for large fragments and should be considered as basic load cases for structural analysis purposes (reference § 29.301). These structural loads should be defined and approved as ultimate loads acting alone. The protective devices and their supporting airframe structures should be able to absorb or deflect the fragment energies defined herein and still continue safe flight and landing. If hazardous, the deflected fragment trajectories and residual energies should also be considered.

(3) Isolation or Redundancy.

(i) Other Engines - Although other engines may be considered critical, engine isolation from rotorburst on multi-engine rotorcraft is not mandatory. Other methods of minimizing the risk to the engine(s) may be acceptable.

(ii) Other Critical Components - Isolation or redundancy of other critical components, the failure of which would not allow continued safe flight and landing should be evaluated relative to the risk of occurrence and where the risk is deemed unacceptable isolation or shielding or other means of reducing the risk should be incorporated.

(4) Composite Materials. If containment devices, shields, or deflectors are chosen by the applicant to be wholly or partially made from composites; they should comply with the structural requirements of AC 20-107A, “Composite Aircraft Structure,” and paragraph AC 29 MG 8, “Substantiation of Composite Rotorcraft Structure,” (which includes glass transition temperature considerations). Glass transition temperature considerations are critical for proper certification of composite or composite hybrid structures used in temperature zones that reach or exceed 200° to 250°F (93° to 121°C) for significant time periods. Hot fragment containment is typically accommodated in such protective devices by use of metal-composite hybrid designs that use the metal component’s properties to absorb the fragment heat load after the entire hybrid structure has absorbed the fragment’s impact load. These devices should comply with §§ 29.609 and 29.1529 to ensure continued airworthiness.

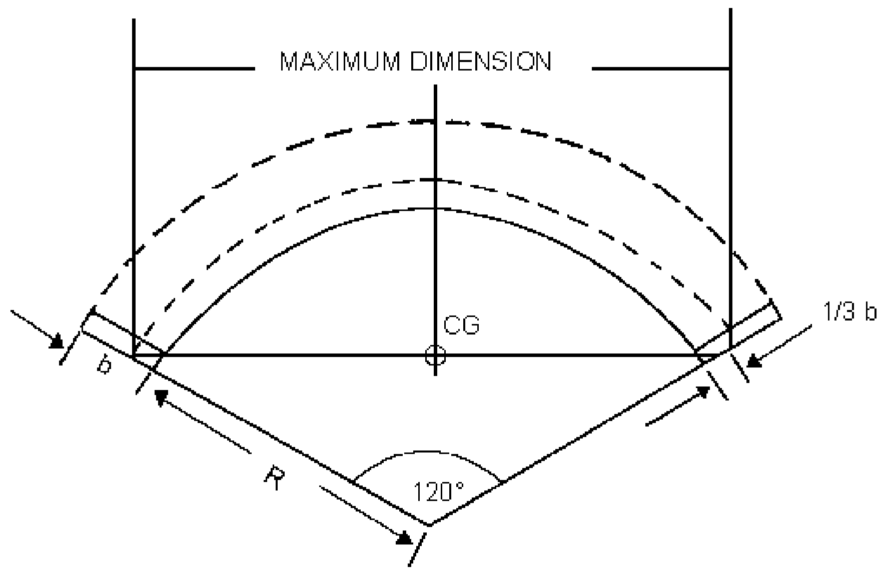


NOTE: 1) THE POSSIBILITY OF TURBINE MOVEMENT SHOULD BE CONSIDERED.

2) ALL ROTORS ARE CONSIDERED TO BE FULLY BLADED FOR CALCULATING MASS.

3) FAILURE OF EACH ROTOR STAGE SHOULD BE CONSIDERED.

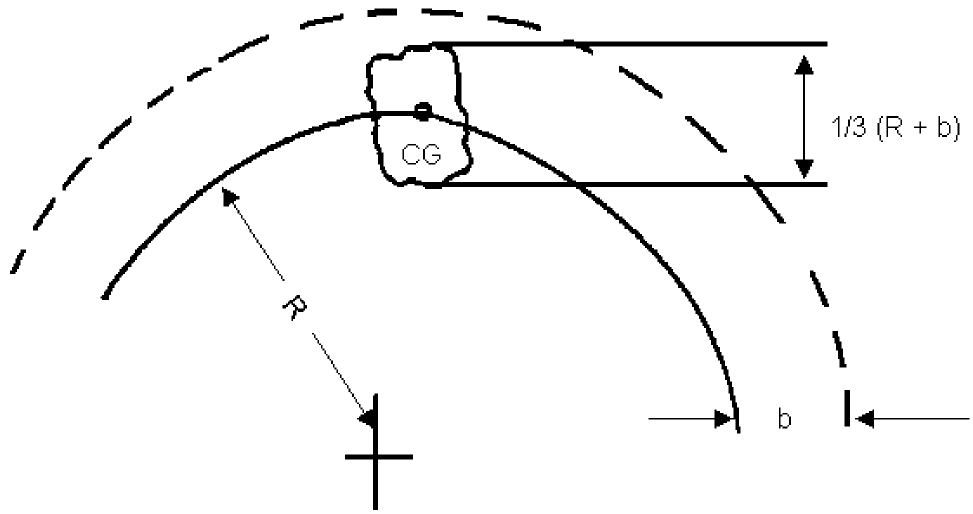
FIGURE AC 29.903C-1 ESTIMATED PATH OF FRAGMENTS



Where R = disc radius
b = blade length

The CG is taken to lie on the maximum dimension as shown.

FIGURE AC 29.903C-2. SINGLE ONE-THIRD DISC FRAGMENT



Where R = disc radius
b = blade length

Maximum dimension = 1/3 (R + b)

Mass assumed to be 1/ 30 th of bladed disc

CG is taken to lie on the disc rim

FIGURE AC 29.903C-3. INTERMEDIATE AND SMALL PIECES OF DEBRIS

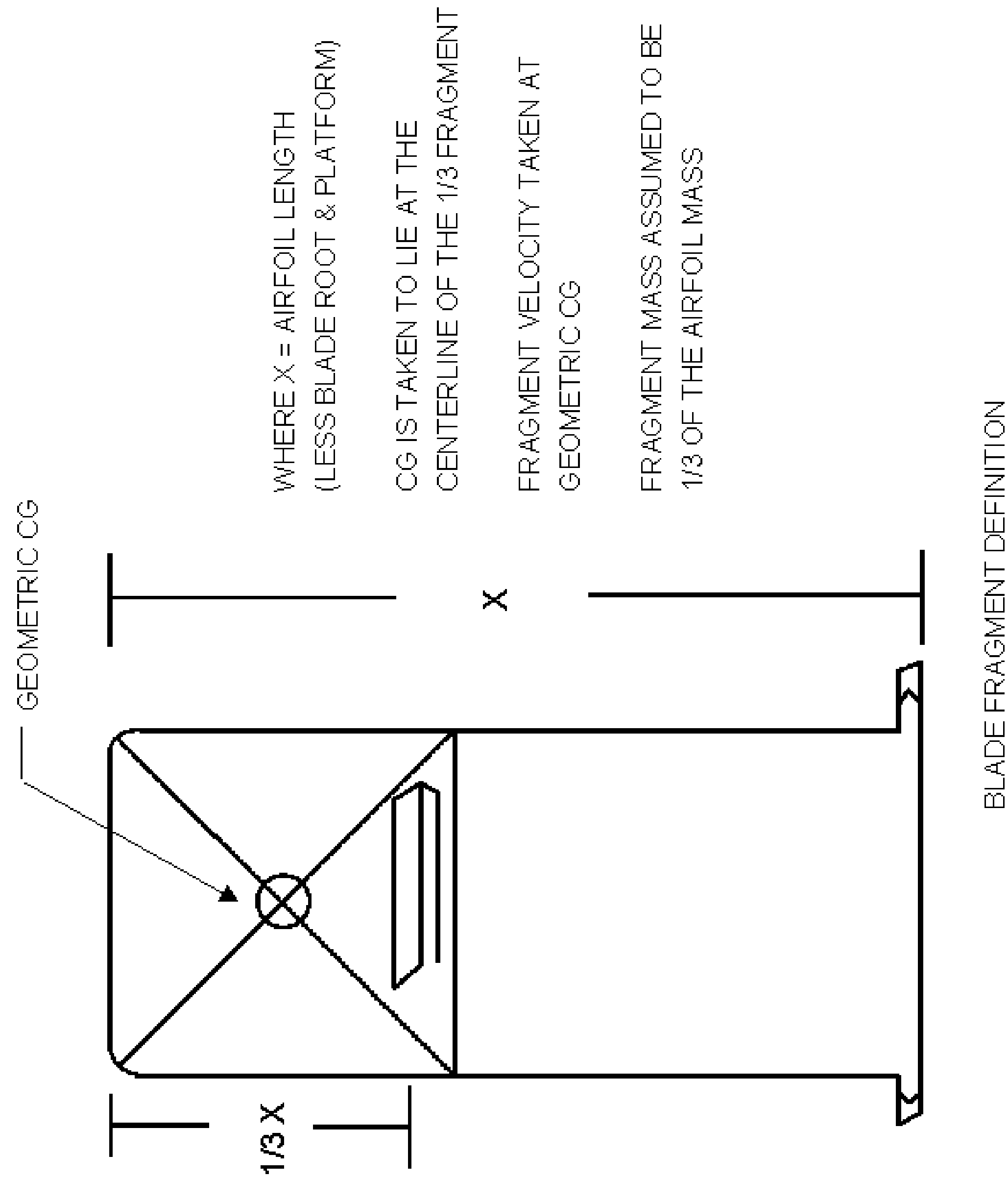


FIGURE AC 29.903C-4

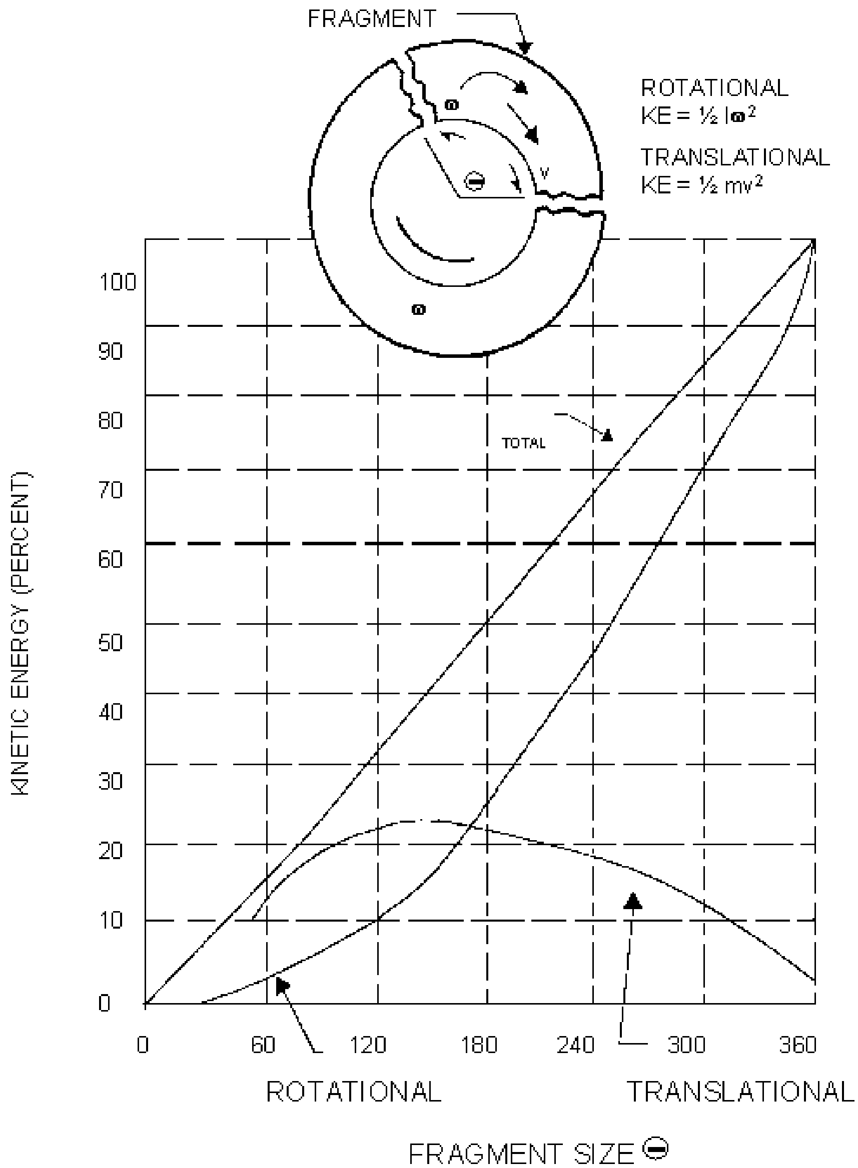


FIGURE AC 29.903C-5 DISTRIBUTION OF TRANSLATIONAL AND ROTATIONAL KINETIC ENERGY OF ROTOR-COMPONENT FRAGMENTS AS A FUNCTION OF FRAGMENT SIZE⊖

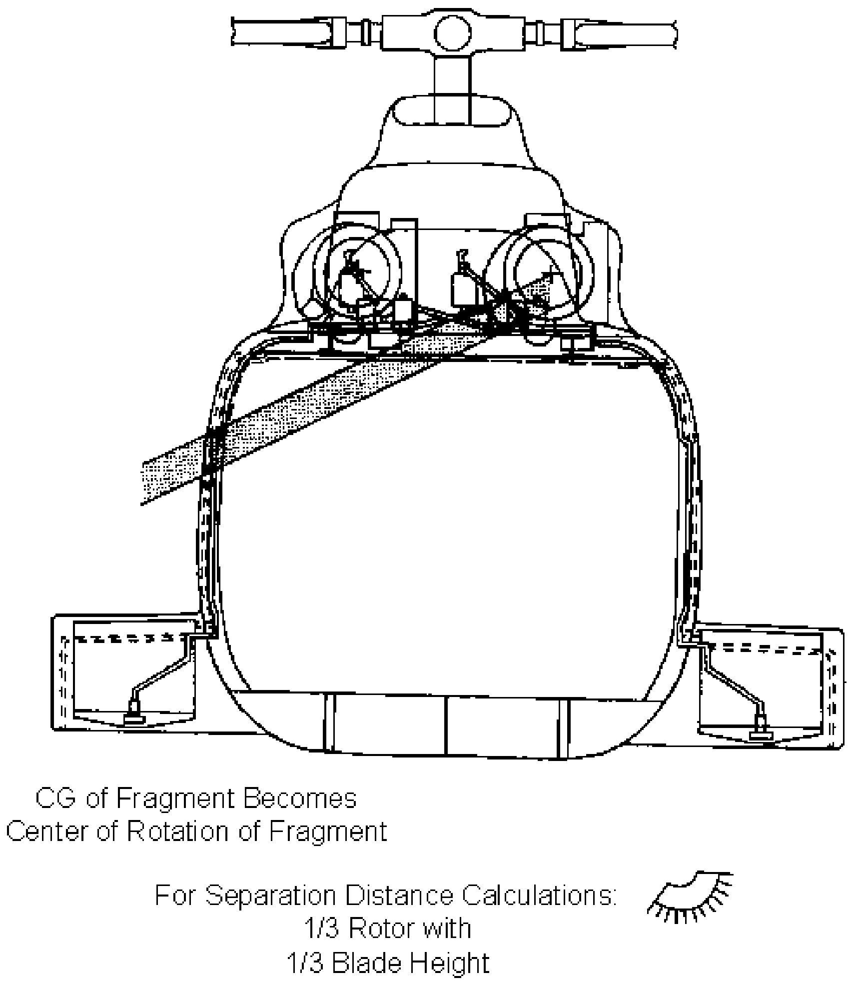


FIGURE 29.903C-6 CROSS SECTION THROUGH AIRCRAFT AT PLANE OF ROTATION OF THE ENGINE DISK FRAGMENT

AC 29.907. <u>§ 29.907 ENGINE VIBRATION</u>.

a. <u>Explanation</u>. This very generalized requirement is authority to require substantiation of the effects of vibration on any part of the engine or the rotorcraft. In normal certification practice, the vibration effects of concern to the powerplant engineer are the vibratory loads or stresses in the engine and in the rotor drive system. Vibration effects on the rotor drive system are of concern if the corresponding loads or stresses result in fatigue damage. This aspect, however, is adequately addressed in § 29.571. Vibration effects on the engine are usually categorized as "installation vibration" and "torsional vibration." Methods of evaluation and limitations of these vibrations are established by the engine manufacturer.

b. <u>Procedures</u>. Review Order 8110.9, Handbook on Vibration Substantiation and Fatigue Evaluation of Helicopter and other Power Transmission Systems. Note that the mechanical coupling of the engines to the rotor drive system creates, for torsional vibration considerations, one, rather complicated, drive system which responds to any forced or resonant frequency. Antinodes or nodes and frequencies may exist in the engine shaft which are absent when the engine is operated on a test stand; therefore, the vibration investigation conducted under Part 33 is not conclusive with respect to torsionals. As noted in Order 8110.9, the engine manufacturers' assistance is necessary to find compliance. Section 29.571 was amended by Amendment 29-13 to include "rotor drive systems between the engines and the rotor hubs" as part of the flight structure. This rule supplements § 29.907 and requires coordination with the structures certification engineer to avoid duplication of effort by the rotorcraft manufacturer. Advisory Circular 20-95, Fatigue Evaluation of Rotorcraft Structure, which provides acceptable methods of compliance with § 29.571, may also be used to find compliance with § 29.907. In addition to basic drive system components such as main and auxiliary rotor drive shafts, the vibratory evaluation should include couplings, gear teeth, gear cases and splines, and should consider, where appropriate, low cycle fatigue associated with ground-air-ground cycles.

AC 29.908. <u>§ 29.908 (Amendment 29-13) COOLING FANS</u>.

a. <u>Section 29.908(a)</u>:

(1) <u>Explanation</u>. This paragraph applies to Category A rotorcraft and is intended to require that powerplant area cooling fans be designed and installed to enable continued safe operation of the rotorcraft after failure of a cooling fan blade. The phrase "except that the loss of cooling need not be considered" at the end of this paragraph is intended to make clear that for the purposes of this section, the FAA/AUTHORITY is concerned only with the fragmentation effect of a fan blade failure (reference Preamble Item 3-64 of Amendment 29-12).

(2) <u>Procedures</u>. If a fan shroud is provided, the applicant may demonstrate that the shroud configuration and strength are adequate to contain a failed fan blade and any other fan blades, guide vanes, etc., which can be expected to fail sequentially

to the initial blade failure. The demonstration can be facilitated by making a saw slot at the root of a blade sufficiently deep to weaken the blade retention strength and create a failure while the fan is rotating at the maximum speed established for the test. If the fan is driven by the rotor drive system, the test speed should be equal to or above the maximum transient speed to be expected with the rotor system. If the fan is driven by other means; i.e., bleed air turbine, hydraulic motor, engine N_1 turbine, etc., the rotational speed for the blade failure demonstration should be based on a critical analysis of speed regimes to be expected. Containment is not required if the fan is located so that blade failure (and any sequential fan component failure) will not jeopardize safety. This may be shown by test or analysis. Segment shielding would likely be involved.

b. Section 29.908(b):

(1) Explanation. This paragraph applies to Category B rotorcraft and is intended to provide safety to the rotorcraft in the event of an assumed cooling fan blade failure or to prescribe a test to show that the cooling fan blade retention means is sufficient that blade failure is not a consideration.

(2) Procedures.

(i) The applicant may select § 29.908(b)(1), (b)(2), or (b)(3) to show compliance with this section. If § 29.908(b)(1) is selected, follow the procedures outlined above for Category A rotorcraft.

(ii) Section 29.909(b)(2) may be selected; however, without containment, damage to any component or structure in the plane of the fan rotor or any other trajectory to be expected should not cause the loss of any function essential to a controlled landing.

(iii) If § 29.908(b)(3) is selected, a spin test at 122.5 percent of the maximum speed associated with either engine terminal speed or an overspeed limiting device would be acceptable to show compliance. No failure should occur, and distortion should not result in fan element contact with housings or other adjacent components. (Note: 150 percent of the centrifugal force is achieved at 122.5 percent of the rotational speed.)

AC 29.908A. § 29.908 (Amendment 29-26) COOLING FANS.

a. Explanation. Amendment 29-26 requires that cooling fans be designed and installed to enable continued safe flight and adequate cooling of the rotorcraft following a fan blade failure. Compliance with the previous requirements could have resulted in hazards to the rotorcraft with the loss of cooling air to critical powerplant components. A new section was also added to the rule for cooling fans, which are not part of the powerplant installation and therefore not subject to the fatigue evaluation under

§ 29.571. It should be determined that no cooling fan blade resonant conditions exist within the operating limits of the rotorcraft unless a fatigue evaluation is conducted.

b. Procedures. Neither mechanical damage nor loss of cooling air should prevent "continued safe flight." The definition of "continued safe flight" is contained in Appendix 1 of AC 20-136 and is quoted as follows:

> Continued safe flight and landing. This phrase means that the aircraft is capable of safely aborting or continuing a takeoff; continuing controlled flight and landing, possibly using emergency procedures but without requiring exceptional pilot skill or strength. Some aircraft damage may occur as a result of the failure condition or upon landing. For airplanes, the safe landing must be accomplished at a suitable airport. For rotorcraft, this means maintaining the ability of the rotorcraft to cope with adverse operating conditions and to land safely at a suitable site.

The FAA/AUTHORITY has determined that for Category A rotorcraft the phrase, "continued safe flight" means that the rotorcraft retains the capability to return and land safely at the point of departure or continue and land safely at the original intended destination or a suitable alternate site.

(1) This section is intended to ensure that a cooling fan blade failure will not jeopardize safety of the rotorcraft. Three ways to show compliance with this section are as follows:

(i) A demonstration should be conducted to show that at the maximum fan speed to be expected, a failed blade will be contained within a housing or shroud which is included in the proposed type design and is designated as the containment shield;

(ii) It should be shown that the installed cooling fan is located such that a blade failure will not jeopardize the safety of the rotorcraft or its ability to continue safe flight (Category A) or land safely (Category B); or,

(iii) It should be shown that the cooling fan blades can withstand an ultimate load 1.5 times the maximum centrifugal force that may be expected in service. The maximum centrifugal forces will occur at the maximum cooling fan rotational speeds. The maximum fan rotational speeds may be related to an overspeed limiting device or to the maximum transient speed to be expected from analysis or test of the engine, system, or component which drives the fan. The maximum rotational speed will be as follows:

(A) For fans driven directly by the engine:

(1) The terminal engine rotational speed that will occur under uncontrolled conditions; such as output shaft disconnect; or

(2) The maximum engine rotational speed that would be controlled by a reliable, approved engine overspeed limiting device.

(B) For fans driven by the rotor drive system, the maximum rotor drive system rotational speed to be expected in service including transients. (Note: Capability to withstand the ultimate load of 1.5 times the centrifugal force means that no failure would occur and distortion should not result in fan element contact with housings or other adjacent components during the 122.5 percent spin test which equates to 150 percent centrifugal force.)

(2) Fatigue. If the cooling fan is not included in the fatigue evaluation under § 29.571, it should be shown that the cooling fan blades are not operating at resonant conditions within the normal operating limits of the rotorcraft.

SUBPART E - POWERPLANT

ROTOR DRIVE SYSTEM

AC 29.917. § 29.917 (Amendment 29-12) DESIGN.

a. Section 29.917(a) General:

(1) Explanation. This paragraph sets forth a definition of the rotor drive system and its associated components. The intent of this paragraph is to clarify and/or establish the identification of components to be considered in other rules which are applicable to the rotor drive system.

(2) Procedures. Coordinate with other certification personnel to ensure that other rules pertaining to rotor drive systems are properly addressed.

b. Section 29.917(b) Arrangement:

(1) Explanation.

(i) Section 29.917(b)(1) pertains to multiengine rotorcraft and requires the drive system arrangement to be such that the rotors will continue to be driven by the remaining engines in order to ensure that lift and control to be expected from the rotors are available if an engine fails.

(ii) Section 29.917(b)(2) pertains to single-engine rotorcraft and is similar to the requirement of paragraph AC 29.917b(1)(i) except that it requires each rotor necessary for operation and control to be driven by the main rotor(s) after disengagement of the engine from the main and auxiliary rotors.

(iii) Section 29.917(b)(3) is intended to require a design which allows the rotor system to be protected from the torsional drag of an inoperative engine.

(iv) Section 29.917(b)(4) pertains to optional torque limiting means (shear sections or clutches) and prohibits these devices from being located in the cross-shafting system between rotors.

(v) Section 29.917(b)(5) is intended to ensure that the design prevents rotors from contacting each other if intermeshing is possible.

(vi) Section 29.917(b)(6) is intended to ensure that locking devices are installed to keep rotors in proper phase if dephasing is possible.

(2) Procedures.

(i) Section 29.917(b)(1) is normally complied with by cross-shafting between rotors, usually via one or more transmissions or gear boxes, to optimize the mechanical simplicity and weight aspects. Individual engine input arrangements are required.

(ii) Section 29.917(b)(2) may be complied with by cross-shafting between rotors. Usually this involves driving the antitorque rotor via a drive shaft from the main transmission.

(iii) Section 29.917(b)(3) may be complied with by installing “free-wheel” or “one-way” clutches in the engine output shaft or transmission input quill. Note that the output section of “free power turbine” engines is not an acceptable method of compliance.

(iv) Section 29.917(b)(4). Any torque limiting devices in the rotor system should be located in the engine output or transmission input quill to ensure that any disconnect from overtorque does not preclude continued normal function and relation of the rotors.

(v) Section 29.917(b)(5). Phase control of intermeshing rotors should utilize positive mechanical drive components. Deflections in both shafting (torsional) and rotors (blade chordwise bending) should be considered in establishing compliance.

(vi) Section 29.917(b)(6). Reconnection of dephased rotors should employ positive mechanical locking pins with secure locking methods.

AC 29.917A. § 29.917 (Amendment 29-40) DESIGN.

a. Explanation. Amendment 29-40 introduces a new § 29.917(b). The previous § 29.917(b) has been redesignated as § 29.917(c). FAR 29.917(a) sets forth a definition of the rotor drive system and its associated components and FAR 29.917(b) requires a design assessment to be performed. The intent of this paragraph (b) is to identify the critical components and to establish and/or clarify their design integrity to show that the basic airworthiness requirements, which are applicable to the rotor drive system, will be met.

b. Procedures.

(1) Section 29.917(a) General. The method of compliance for this section is unchanged.

(2) Section 29.917(b) Design Assessment. A design assessment of the rotor drive system should be carried out in order to substantiate that the system is of a safe design and that compensating provisions are made available to prevent failures classified as hazardous and catastrophic in the sense specified in paragraph (c) below. In carrying out the design assessment the results of the certification ground and flight

testing (including any failures or degradation) should be taken into consideration. Previous service experience with similar designs should also be taken into account (see also FAR 29.601(a)).

c. Definitions. For the purposes of this assessment, failure conditions may be classified according to the severity of their effects as follows:

(1) Minor. Failure conditions which would not significantly reduce rotorcraft safety, and which involve crew actions that are well within their capabilities. Minor failure conditions may include, for example, a slight reduction in safety margins or functional capabilities, a slight increase in crew workload, such as routine flight plan changes, or some inconvenience to occupants.

(2) Major. Failure conditions which would reduce the capability of the rotorcraft or the ability of the crew to cope with adverse operating conditions to the extent that there would be, for example, a significant reduction in safety margins or functional capabilities, a significant increase in crew workload or in conditions impairing crew efficiency, or discomfort to occupants, possibly including injuries.

(3) Hazardous. Failure conditions which would reduce the capability of the rotorcraft or the ability of the crew to cope with adverse operating conditions to the extent that there would be -

(i) A large reduction in safety margins or functional capabilities;

(ii) Physical distress or higher workload such that the flight crew cannot be relied upon to perform their tasks accurately or completely;

(iii) Serious or fatal injury to a relatively small number of the occupants;

(iv) Loss of ability to continue safe flight to a suitable landing site.

(4) Catastrophic. Failure conditions which would prevent a safe landing.

(5) Minimize. Reduce to the least possible amount by means that can be shown to be both technically feasible and economically justifiable.

(6) Health Monitoring. Equipment, techniques, and/or procedures by which selected incipient failure or degradation can be determined.

d. Failure Analysis.

(1) The first stage of the design assessment should be the Failure Analysis, by which all the hazardous and catastrophic failure modes are identified. The failure analysis may consist of a structured, inductive bottom-up analysis, which is used to evaluate the effects of failures on the system and on the aircraft for each possible item

or component failure. When properly formatted it will aid in identifying latent failures and the possible causes of each failure mode. The failure analysis should take into consideration all reasonably conceivable failure modes in accordance with the following:

(i) Each item/component function(s).

(ii) Item/component failure modes and their causes.

(iii) The most critical operational phase/mode associated with the failure mode.

(iv) The effects of the failure mode on the item/component under analysis, the secondary effects on the rotor drive system and on the rotors, on other systems and on the rotorcraft. Combined effects of failures should be analyzed where a primary failure is likely to result in a secondary failure.

(v) The safety device or health monitoring means by which occurring or incipient failure modes are detected, or their effects mitigated. The analysis should consider the safety system failure.

(vi) The compensating provision(s) made available to circumvent or mitigate the effect of the failure mode (see also paragraph (1) below).

(vii) The failure condition severity classification according to the definitions given in paragraph (c) above.

(2) When deemed necessary for particular system failures of interest, the above analysis may be supplemented by a structured, deductive top-down analysis, which is used to determine which failure modes contribute to the system failure of interest.

(3) Dormant failure modes should be analyzed in conjunction with at least one other failure mode for the specific component or an interfacing component. This latter failure mode should be selected to represent a failure combination with potential worst case consequences.

(4) When significant doubt exists as to the effects of a failure, these effects may be required to be verified by tests.

e. <u>Evaluation of Hazardous and Catastrophic Failures</u>.

(1) The second stage of the design assessment is to summarize the hazardous and catastrophic failures and appropriately substantiate the compensating provisions which are made available to minimize the likelihood of their occurrence. Those failure conditions that are more severe should have a lower likelihood of occurrence

associated with them than those that are less severe. The applicant should obtain early concurrence of the cognizant certificating authority with the compensating provisions for each hazardous or catastrophic failure.

(2) Compensating provisions may be selected from one or more of those listed below, but not necessarily limited to this list.

(i) Design features; i.e., safety factors, part-derating criteria, redundancies, etc.

(ii) A high level of integrity.

(iii) Fatigue tolerance evaluation.

(iv) Flight limitations.

(v) Emergency procedures.

(vi) An inspection or check that would detect the failure mode or evidence of conditions that could cause the failure mode.

(vii) A preventive maintenance action to minimize the likelihood of occurrence of the failure mode, including replacement actions and verification of serviceability of items which may be subject to a dormant failure mode.

(viii) Special assembly procedures or functional tests for the avoidance of assembly errors which could be safety critical.

(ix) Safety devices or health monitoring means beyond those identified in paragraphs (vi) and (vii) above.

AC 29.921. <u>§ 29.921 ROTOR BRAKE</u>.

a. <u>Background</u>. Rotor brake safety requirements are intended not only to prevent adverse effects on aircraft performance due to brake drag but also to minimize the possibility of fires. These fires, caused by friction from a dragging rotor brake, have occurred both in flight and during ground operation with extremely hazardous consequences.

b. <u>General</u>. This rule requires (1) that any limitations on the use of the rotor brake must be established, and (2) that the control for the brake must be guarded to prevent inadvertent operation.

c. <u>Limitations</u>.

(1) The limitations on the use of the rotor brake should first be defined by the applicant and will normally consist of merely the maximum rotor speed eligible for application of the brake. In some installations, limitations associated with engine operation may be specified. For example, some “free power section” type turbine engines can be safely operated within certain low limits with the rotor brake engaged, while other engines cannot tolerate this condition. At least one manufacturer has included a maximum rotor speed for emergency rotor brake application. This is considered an enhancing safety consideration and is recommended.

(2) Control guard mechanisms to prevent inadvertent operation may be conventional. A cockpit evaluation should be conducted by flight test personnel to affirm the function of the guard and the brake, and that markings, if any, are adequate and that both latched and unlatched positions of the control do not interfere with other cockpit functions.

d. General qualification aspects should include:

(1) The 400 applications required by § 29.923(j) conducted as a part of the § 29.923 endurance test.

(2) Torsional vibration measurements of the loads in the brake components and the rotor drive system during a critical brake engagement procedure, with appropriate consideration in the fatigue evaluation for these components. Brake engagements should be conducted with and without collective control displacement as authorized by the flight manual or a training manual.

(3) Brake component temperature measurements during a critical brake application in conjunction with an evaluation of the general brake compartment for compliance with §§ 29.863 and 29.1183.

(4) Placards, decals, and flight manual limitations and instructions appropriate to operate the rotor brake safely.

(5) An evaluation for hazardous failure modes as required by § 29.901(c). If the brake hydraulic system is integral with the rotorcraft hydraulic system, failure modes of pressure regulators and control valves, including valve leakage, will be of interest. Mechanical cams, calipers, and levers may be prone to seize or fail to release the brake due, in part, to corrosion and lack of lubrication to be expected when brake components encounter high temperature cycling.

NOTE: Most rotor brakes include nonmetallic pucks or liners, usually included in nonrotating brake components, which are subject to wear in proportion to the number of applications. Replacement of these pucks during the § 29.923 endurance test has been found acceptable provided the reason for replacement is simply wear and not because of any change in brake loading, disk temperature, or vibratory characteristics which can be expected in service. Verify that the maintenance manual includes a routine check for excessive puck or liner wear.

e. Other comments. Rotor brakes may be added to the basic design as a postcertification program without necessarily reconducting the complete § 29.923 endurance test provided:

(1) Steady and vibratory stresses in brake components, the rotor drive system, and in the rotor system itself are determined and shown to be acceptable.

NOTE: Moments, stresses, etc., from brake operation apply loads to the drive system in the reverse direction to normal powered flight. Advise the airframe engineer to require evaluation of chordwise bending loads in the hub and blade components of the main rotor system.

(2) The 400 brake engagements of § 29.923(j) should be accomplished with a complete rotor and rotor drive system, followed by disassembly sufficient to verify that all components subject to loads from the brake remain serviceable. Since this test may be so short as not to cause appreciable wear patterns to appear, special pretest coatings such as black oxide or Du-Lite may be needed on gear teeth and bearing races to distinguish and evaluate the contact patterns. Information on maximum deceleration rates should be supplied to the manufacturer of the engines to be used in the rotorcraft for evaluation of the acceptability of backloading or motoring of turbines, fuel control components, torque meters, etc.

AC 29.923. § 29.923 (Amendment 29-17) ROTOR DRIVE SYSTEM AND CONTROL MECHANISM TESTS.

a. Explanation.

(1) This rule is intended to require demonstration that the rotor drive system, as defined in § 29.917(a), is capable of normal operation within the limitations proposed, without hazard of failure from excessive wear or deterioration due to mechanical loads. The basic test is not designed and should not be expected to demonstrate safety from oscillatory stresses normally investigated under §§ 29.571 and 29.907, although any data generated by these tests, which are in fact applicable to showing compliance with §§ 29.571 and 29.907, may be used. Some variations in the endurance test plan to generate data applicable to the vibration substantiative effort or other qualification aspects may be acceptable if the basic requirements of the endurance test are preserved.

(2) The construction of this rule is such that a series of runs, each at least (but not limited to) 10 hours in length must be repeated 20 times, for a total of (at least) 200 hours of test, not including time required to adjust power or to stabilize operating conditions for those conditions that require stabilization. Extension of the total test beyond 200 hours (or extension of test runs beyond 10 hours) will occur if qualification

for the 2½-minute one-engine-inoperative (OEI) optional rating is proposed by the applicant. The 30-minute OEI rating qualification test will extend the test beyond 200 hours for rotorcraft equipped with three or more engines. Also, compliance with § 29.923(g) may result in extended endurance tests if dynamic or malfunction conditions exist which adversely affect the endurance tolerance of the rotor drive system. Section 29.923(a) should be interpreted as requiring test runs or cycles to be repeated in essentially the same sequence, although more than 10 hours may be needed to complete a run or cycle. This section also requires the test to be conducted "on the rotorcraft." This means a rotorcraft in conformity to the design for which approval is requested. However, many nonconformity features, such as doors, some cowling and instrumentation, fuel tanks (alternate external fuel supply may be utilized), interior features, fire detectors, extinguishers, inlet ducts, exhaust baffles, etc., may be acceptable provided each item is technically considered and found to be unimportant to the test results. Any significant deviations from the conformed rotorcraft configuration should be coordinated with the cognizant FAA/AUTHORITY engineering staff and if found acceptable, documented as such. The restraint (tie-down) arrangement used during the test will necessarily be arranged to react rotor thrust loads in lateral as well as vertical directions. However, the restraint should permit normal deflections due to rotor thrust in the engine and drive system support arrangement.

(3) Safety cables may be installed normal to the tail boom at the tail rotor gearbox location; however, restraint may be provided to keep airframe deflections from exceeding those expected in normal and accelerated flight.

(4) The test torque requirements of § 29.923(a)(3)(i) mean the torque values for which approval is requested but not to exceed the values approved for the respective limits for the engine to be used. However, an applicant should be allowed to qualify the rotor drive system for torque values higher than those for which approval is requested if the engines actually used are capable of the torque and can be shown by an output shaft torsional investigation to be equivalent or conservative with respect to torsional vibration to the engines proposed for the initial certification configuration. Variations in rotational speed from the certification values should not be allowed except where careful evaluation of vibration aspects, bearing loads, centrifugal stiffening effects, and torque variations are conducted.

(5) The rotor configuration required by § 29.923(a)(3)(ii) is intended to assure that lift, torque, and vibration loads to be expected in service are introduced into the endurance test, although the presence of the vibration aspects does not normally satisfy the vibration evaluations required by §§ 29.571 and 29.907. In fact, vibration modes may be changed and amplified by the tie-down restraints and the increased thrust to be expected from in-ground effects on the rotor system. These effects, although unquantified, are intended as a normal part of endurance testing. Preproduction rotor blades have been successfully used in endurance tests but only after specific investigations of blade properties such as stiffness, inertia and inertia distribution, thrust and blade bending, and torsional frequency response have been carefully compared to assure validity of the test. The endurance test includes testing of

the rotor control mechanism. Conformity of the rotors may be very significant to this aspect of the test.

b. Procedures.

(1) Section 29.923(b)(1) prescribes the takeoff portion of the endurance test. This test involves a series of 5-minute repetitive runs at the torque and at the engine/rotor rotational speed selected by the applicant for the takeoff limit for the rotorcraft. These values of torque (manifold pressure, for reciprocating engines) and RPM should correspond to the red radials on the corresponding powerplant instruments, except on installations where uncompensated engine governor “droop” results in a higher rotational speed for lower powers. The requirement in this section for declutching the engine may be difficult to achieve if engine deceleration and rotor system deceleration rates are similar. In some cases, the engine fuel control deceleration schedule may be adjusted to achieve clutch disengagement, otherwise, an engine shaft brake mechanism may be needed.

(2) The torque and speed requirements for the optional 2½-minute one-engine-inoperative (OEI) tests should be interpreted as described above for the takeoff runs. If the test is conducted during warm ambient conditions, excessive engine gas temperatures may be required to achieve the torque and speed conditions required by this part of the test. Minor adjustments in the run schedule may be allowed to take advantage of cooler nighttime ambient temperatures. Addition of water/alcohol systems to increase engine hot-day power may be appropriate in some instances. Liquid nitrogen spray into engine inlets has also been used to depress inlet temperatures sufficiently to obtain test conditions.

(3) In § 29.923(c), (d), (e), and (f), the torque requirements should be interpreted as above; i.e., the run should be made with maximum continuous torque or percentage thereof, as specified by the subparagraph, and the rotational speed should be maximum continuous for paragraph (d) and the lowest permissible “power-on” speed for paragraphs (e) and (f). Rotor control cycling must be accomplished during the “maximum continuous” portion of the endurance run. The controls of concern are the flight controls; i.e., cyclic and directional controls for rotorcraft with tail rotor and single main rotor. The collective control is normally used to set power and is not involved in control cycling. During control cycling the controls may be cycled from stop to stop, or a limited travel may be accepted if the travel produces the maximum fore and aft, left and right, and yaw thrust components of the rotors as measured in flight. One method of determining the required control displacement is to measure main rotor mast bending in level forward flight at maximum continuous power for the forward control displacement limit, and in level rearward flight at maximum continuous power (or the power associated with the maximum rearward flight speed to be expected) for the aft control displacement limit. Using the same mast bending instrumentation, with the rotorcraft in the ground tie-down situation, and with collective control set for maximum continuous power, displace the cyclic fore and aft to obtain the same mast bending as measured in flight. Similar measurements and control displacements may be used for sideward

thrust components. Yaw control displacement should consider maneuver requirements in conjunction with sideward flight. Critical gross weight and center of gravity should be used to establish test conditions. These same procedures may be used to establish limited control positions required to comply with § 29.923(i) except that typical flight conditions to be used would be stabilized level flight at maximum continuous power, climb at maximum continuous power, and hovering, including stabilized sideward and rearward flight. Note that for § 29.923(i)(1) vertical thrust is required. Depending on the mast angle and center of gravity, this condition may not necessarily involve zero mast bending loads. Vertical thrust may be used during the takeoff run, including the runs at 2½-minute power and the overspeed run of § 29.923(h). One-engine-inoperative runs (§ 29.923(k)) should be conducted with the cyclic set for maximum forward thrust. For these runs and any run that does not specify the position for the yaw control, that control should be set to react main rotor torque.

(4) Section § 29.923(g) provides for introducing special tests into the endurance tests to demonstrate that the transmission and drive system can tolerate certain engine malfunctions to be expected. This was originally directed at demonstrating safety in the event of spark plug or magneto failures of reciprocating engines. Turbine engines normally do not exhibit failure modes suitable for substantiation by endurance testing; however, severe or abusive operating conditions which must be expected to occur in service should be defined and included in this test. Conditions or phenomena to be considered should include but not be limited to moderate engine surge, abusive clutch engagements, torque mismatching, anticipated control mishandling, and so forth. Alternatively, repeating the takeoff run of § 29.923(b) may be appropriate. It is not intended that the special testing for 2½-minute power be repeated if a rerun of the takeoff power run is required by § 29.923(g).

(5) Section 29.923(h) requires overspeed testing at the torque which will produce maximum continuous power and at the maximum rotational speed to be expected. Normally this would be the maximum transient, power-on rotor speed available with speed controls operating. Special control adjustments for test purposes may be needed to achieve the required test conditions.

(6) Section 29.923(i) requires stabilized flight control displacement according to a prescribed schedule. The control displacement should be the same as derived to show compliance with § 29.923(c)(2).

(7) Section 29.923(j) requires 400 clutch and brake engagements. These tests are prescribed to establish a level of reliability of clutch and brake components installed as a part of the rotor drive system of rotorcraft. The clutch tests apply to all clutches installed to comply with § 29.917(b)(3), and each such clutch must be tested. A rotor brake is not required for certification, although a brake of some type may be installed temporarily to facilitate conducting the clutch testing required by this section. Clutch disengagement is also required by this section, thus, malfunction of the disengagement feature would be a basis for discontinuance. Some rotorcraft configurations (those with single-spool turbine engines or reciprocating engines) include an additional clutch to

decouple the engine from the drive system to facilitate engine starting. These clutches should also be exercised at least 400 times during this test.

(8) Section 29.923(k) sets forth the optional tests to be conducted if a 30-minute OEI rating is requested. It may be noted that the time for conducting this test replaces time deducted from the run of § 29.923(f). Flight control positions should be set for level flight or climb, whichever produces the maximum forward thrust component, and the antitorque system control should be set to react the maximum rotor torque. The torque and rotational speed values should be the maximum for which approval is requested.

(9) Section 29.923(m) normally is satisfied by the requirements of §§ 29.571 and 29.907.

(10) Section 29.923(n) requires special tests for rotor drive systems designed to operate at two or more gear ratios. Depending on the limitations and instructions proposed for operating at other gear ratios, additional tests (beyond the normal 200-hour schedule) or substitutions into the basic test should be conducted to qualify the rotor drive system for operations at other gear ratios. The length of testing, torque and speed requirements, overspeed tests, and control positions for these tests should parallel the requirements of the basic endurance test.

(11) Section 29.923(o) requires the rotor drive system and rotor control mechanism to be in a serviceable condition at the end of the test. Verification of this requirement requires a complete disassembly and examination of the entire rotor drive system and rotor control mechanism. The disassembly itself should be closely monitored for evidence of adequate breakaway torque on all bolted fasteners. Samples of lubrication from oil sumps and filters should be retained for spectrographic analysis, and seals should be examined for possible damage due to test requirements. Care should be taken to differentiate between seal damage and bearing damage due to disassembly procedures so that the direct results of the test may be properly considered. Close visual observation of each tooth on each gear is necessary to affirm proper load/contact patterns and absence of excessive surface stress or scrubbing motions. Bearings should be examined to verify that ball or roller paths are within limits, bearing cages are undamaged, and bearing balls or rollers and their races are free from pitting. Any evidence of bearing races turning or spinning in respective housing or bores probably indicates design or fit deficiencies. The applicant should have available wear limits data which include items such as distance across pins and tooth profile limits for gears. Many of these items require special, close tolerance inspection equipment and trained inspectors to determine compliance. In some instances bearings, clutches, oil pumps, etc., should be returned to the original manufacturer for a finding of serviceability. Localized overheating, usually exhibited by discolorations is an indication of an unsatisfactory condition. Should any of the items discussed above or other defects appear such that the component is unserviceable, a redesign which includes recognizable improvements should be required before authorizing a retest. To simply “try again” in hopes of success should not be accepted.

(12) This section also prohibits intervening disassembly which might affect test results. Generally, this simply means no disassembly whatsoever. However, some very limited disassembly can usually be conducted provided care is used to assure that items such as critical fastener torques or gear backlash controls are not disturbed.

c. Additional Test Considerations.

(1) Pressure Lubricated Gearboxes. The endurance test hardware can be adjusted/modified to sustain high-limit oil temperature and low-limit oil pressure in order to provide a basis for approval of the values listed as limits. A minimum of 20 hours at maximum continuous torque and maximum continuous rotational speed should be involved in the test. Other parameters such as minimum oil temperature and maximum oil pressure may more appropriately be evaluated by bench test. The significant points here are effects of extremely high oil pressure (due to the high viscosity of cold oil) on any positive displacement oil pump, on filters for possible collapse, on oil coolers for possible rupture due to internal pressure, seals, bypass valves, and most important, adequate lubrication of gears, bearings, etc., under conditions of minimal oil flow. Normally, an operation restriction against exceeding idle power/speed conditions until significant warm-up occurs is prescribed. Individual component qualification tests may provide data to meet some of these aspects.

(2) The existing endurance test schedule does not necessarily provide for any asymmetric power inputs from multiengine drive system arrangements. For this situation, the drive system should at least be subjectively evaluated for possible hazards or excessive loads to be expected from asymmetric torque inputs and additional testing prescribed under the authority of § 29.923(g). The extent and severity of these tests should be established in consideration of the design peculiarities, the recommended operating procedures, and any OEI tests included in this test schedule.

(3) Accessory Drives. Normally, all accessory drives on a gearbox will be loaded during the endurance test. Electrical load banks or other suitable methods may be used to assure that the generator drives are loaded and thus properly qualified. Hydraulic pumps may be loaded by resetting hydraulic system relief valves to maintain limit pressure (load) continuously. If this condition is excessively severe, a method of load cycling may be appropriate. Note that accessory loads reduce the power available to the main rotor. Also, tail rotor loads are, insofar as the transmission is concerned, another large accessory. Care should be taken to assure that in-flight unloading of these accessory drives, including the tail rotor does not subject the main gearbox to loads significantly beyond those qualified by endurance tests.

(4) Gearbox Oil Tanks. Normally, gearbox oil is contained in an integral cast sump which, for other reasons, has sufficient strength to obviate the need for pressure tests. However, a subjective evaluation should be made to assure that detail design features such as sight gauges, filler caps, etc., offer adequate strength.

AC 29.923A. § 29.923 (Amendment 29-26) ROTOR DRIVE SYSTEM AND CONTROL MECHANISM TESTS.

a. Explanation. Amendment 29-26 includes additional endurance test criteria for a new continuous OEI rating, and clarifies the torque and RPM relation intended for the various power ratings involved in the tests prescribed by this section.

(1) Section 29.923(a)(1) was amended to require that the test cycle be extended beyond 10 hours if OEI rating tests are included in the test program. This change was needed to maintain the cycle aspect of the test if OEI ratings are included.

(2) Section 29.923(a)(3) was amended to include rotational speed as a part of the test because the term “torque” by itself does not adequately define the test requirements.

(3) Section 29.923(b)(2), (f), and (k) were amended to add the test requirements for the new continuous OEI rating and retain, as an alternate, the 30-minute OEI rating tests for those applicants who may request this rating. This change provided a regulatory test basis for qualifying the rotor drive system for optional OEI ratings.

(4) Section 29.923(g) was amended to remove the inference that the 2½-minute OEI runs should be repeated if the takeoff run is reconducted. Under these circumstances, additional testing for the 2½-minute rating is unnecessary.

b. Procedures.

(1) The construction of this amendment is such that a series of runs, each at least (but not limited to) 10 hours in length, should be repeated 20 times for a total of at least 200 hours of test. The time required to adjust power or to stabilize operating conditions for those conditions that require stabilization is not included. Figure AC 29.923A-1 shows a graphic representation of the 10-hour test cycle. Extension of the total test time beyond 200 hours (or extension of test runs beyond 10 hours) will occur if qualification for the 2½-minute, 30-minute, or continuous OEI optional ratings is proposed by the applicant for rotorcraft equipped with two or more engines. Also, compliance with § 29.923(g) may result in extended endurance tests if dynamic or malfunction conditions exist which adversely affect the endurance tolerance of the rotor drive system. Section 29.923(a) should be interpreted as requiring test runs or cycles to be repeated in essentially the same sequence, although more than 10 hours may be needed to complete a run or cycle. This section also requires the test to be conducted “on the rotorcraft.” This means a rotorcraft that is in conformity to the type design for which approval is requested. However, many nonconforming features, such as doors, some cowling and instrumentation, fuel tanks (alternate external fuel supply may be utilized), interior features, fire detectors, extinguishers, inlet ducts, exhaust baffles, etc., may be acceptable provided each item is technically considered and found to have no impact on the test results. Any significant deviations from the

conformed rotorcraft type design should be coordinated with the cognizant FAA/AUTHORITY engineering staff and, if found acceptable, properly documented. The restraint (tie-down) arrangement used during the test should be arranged to react rotor thrust loads in lateral as well as vertical directions. The restraint should permit normal deflections due to rotor thrust in the engine and drive system support arrangement.

(2) The test torque and speed requirements of § 29.923(a)(3)(i) refer to the torque/speed combination (or power) values for which approval is requested. The requested torque/speed combination should not exceed the limits approved for the respective engine(s) to be used. An applicant may qualify the rotor drive system for torque values higher than those approved for the engine.

(3) In §§ 29.923(c), (d), (e), and (f), the torque requirements should be interpreted as above; i.e., the run should be made with maximum continuous torque or percentage thereof, as specified by the subparagraph; and the rotational speed should be maximum continuous for paragraph (d) and the lowest permissable “power-on” speed for paragraphs (e) and (f). Rotor control cycling should be accomplished during the “maximum continuous” portion of the endurance run. The controls of concern are the flight controls (cyclic and directional controls for typical rotorcraft). The collective control is normally used to set power and is not involved in control cycling. During control cycling, the controls may be cycled from stop to stop; or a limited travel may be accepted if the travel produces the maximum fore and aft, left and right, and yaw thrust components of the rotors as measured in flight. The frequency for cycling the controls is defined in §§ 29.923(c)(1), (2), and (3), and specified in Note 3 of figure AC 29.923A-1. One method of determining the required control displacement is to measure main rotor mast bending in level forward flight at maximum continuous power for the forward control displacement limit, and in level rearward flight at maximum continuous power (or the power associated with the maximum rearward flight speed to be expected) for the aft control displacement limit. Using the same mast bending instrumentation, with the rotorcraft in the ground tie-down situation, and with collective control set for maximum continuous power, displace the cyclic fore and aft to obtain the same mast bending as measured in flight. Similar measurements and control displacements may be used for sideward thrust components. Yaw control displacement should consider maneuver requirements in conjunction with sideward flight. Critical gross weight and center of gravity should be used to establish flight test conditions. These same procedures may be used to establish limited control positions required to comply with § 29.923(i), except that typical flight conditions to be used would be stabilized level flight at maximum continuous power, climb at maximum continuous power, hover, and stabilized sideward and rearward flight. Note that for § 29.923(i)(1), vertical thrust is required. Depending on the mast angle and center of gravity, this condition may not necessarily involve zero mast bending loads. Vertical thrust may be used during the takeoff run, including the runs at 2½-minute power and the overspeed run of § 29.923(h). One-engine-inoperative runs (§ 29.923(k)) should be conducted with the cyclic set for maximum forward thrust. For these runs and any run

that does not specify the position for the yaw control, that control should be set to react to main rotor torque.

(4) Section 29.923(k) sets forth the optional tests to be conducted if a 30-minute or a continuous OEI power rating is requested. Flight control positions should be set for level flight or climb (whichever produces the maximum forward thrust component) and the anti-torque system control should be set to react the maximum rotor torque. The torque and rotational speed values should be the maximum for which approval is requested.

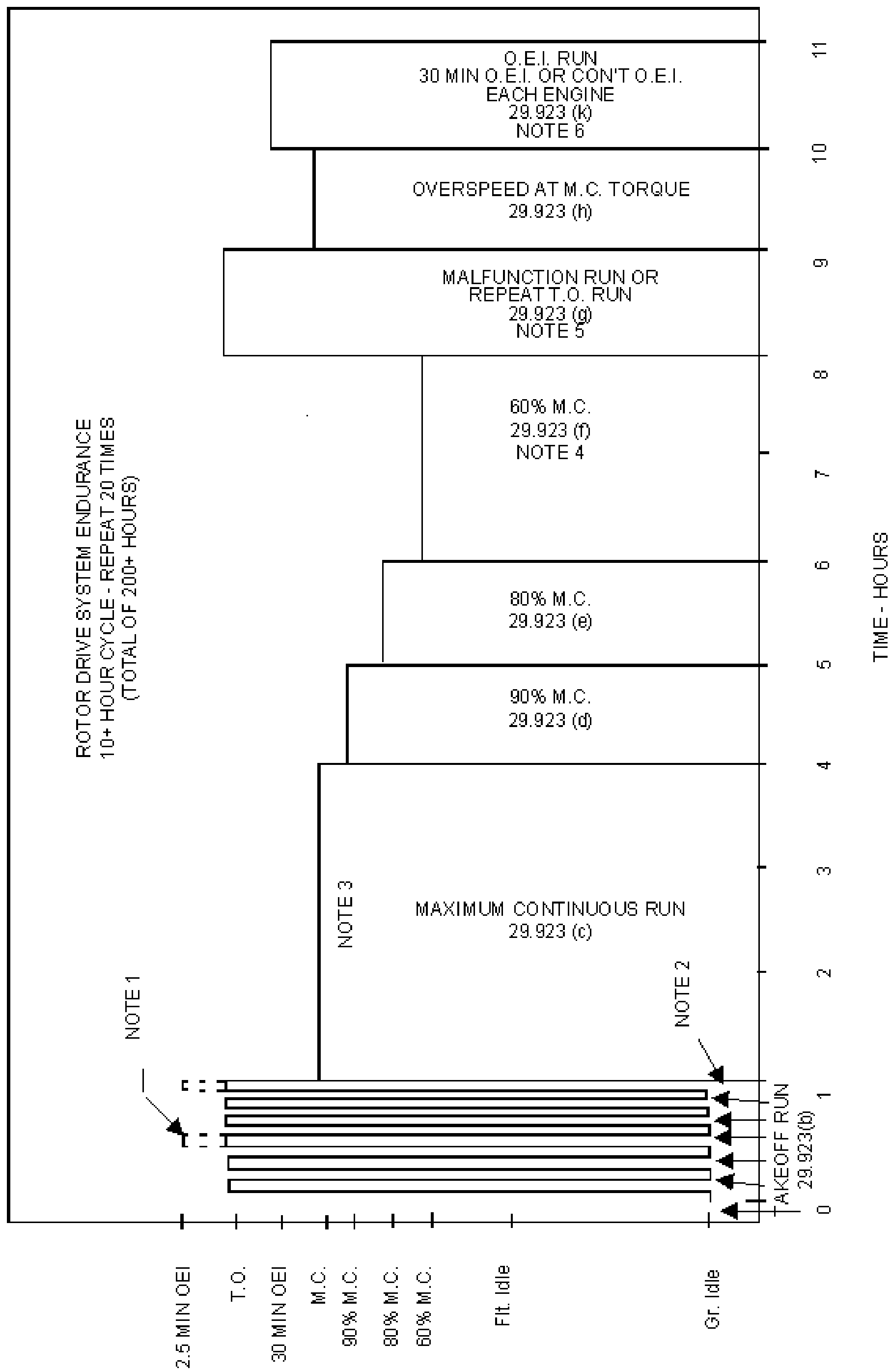


FIGURE AC 29.923A-1

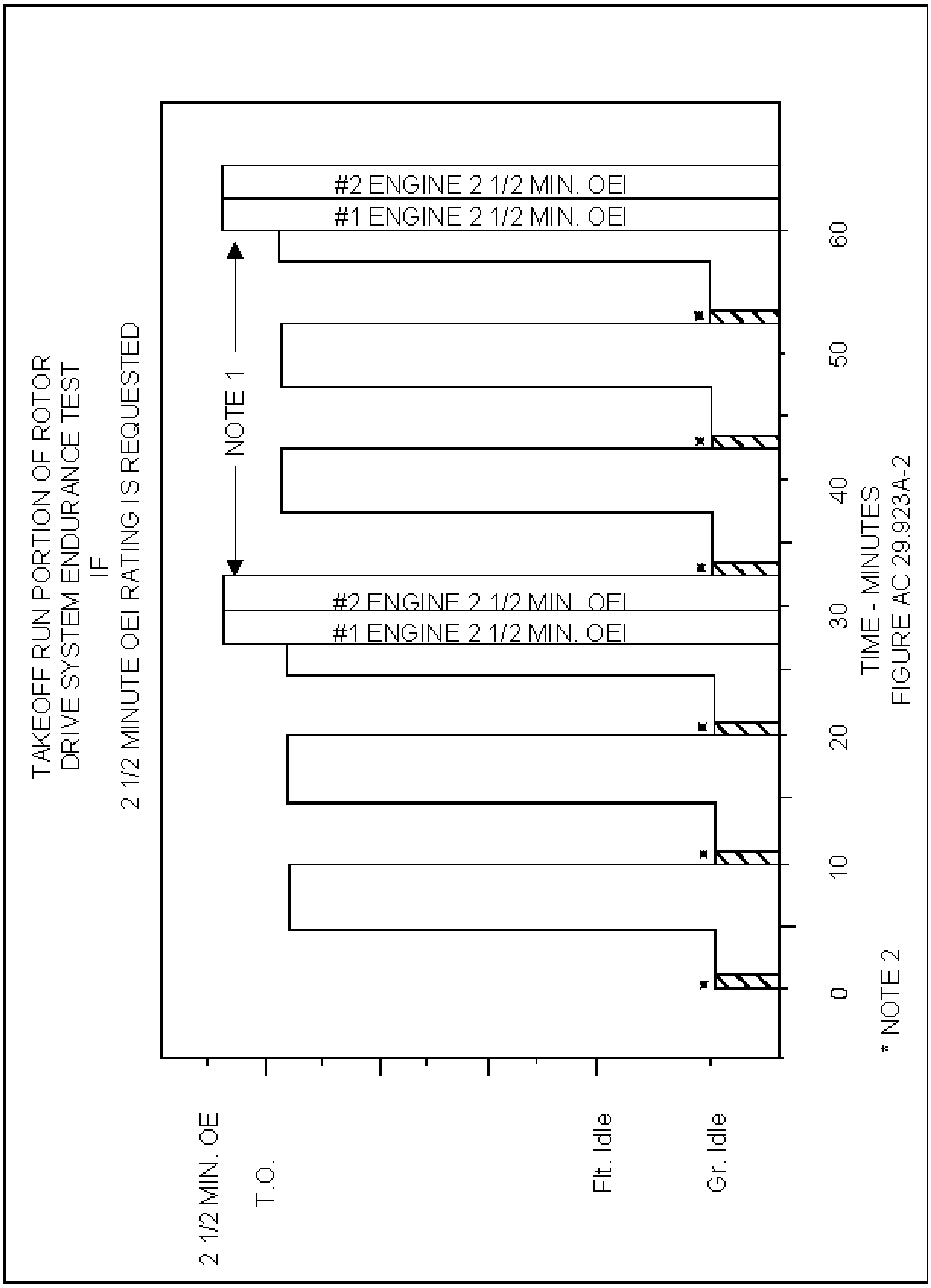


FIGURE AC 29.923A-2

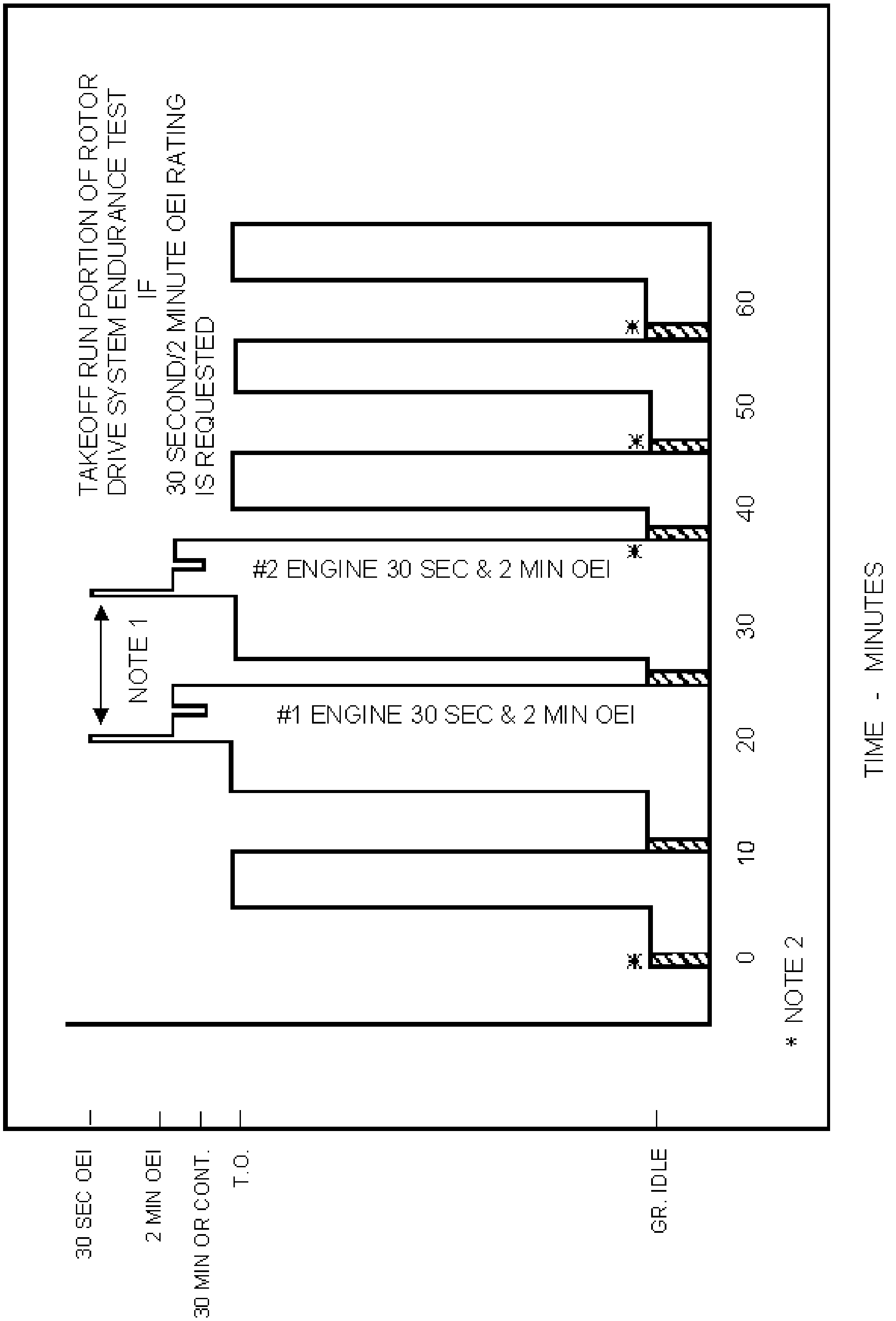


FIGURE AC 29.923A-3

Figures AC 29.923A-1, AC 29.923A-2, and AC 29.923A-3 Notes

1. If the 2½-minute OEI rating is requested, the following should be conducted for each engine: Demonstrate 2½-minute OEI power twice or 5 minutes per cycle for a total of 100 minutes at 2½-minute OEI power during the 200-hour endurance test. See figure AC 29.923A-2 for a graphic description of the takeoff run for a two-engine rotorcraft using the 2½-minute OEI rating (Refer to § 29.923(b)(2)). If the 30-second/2-minute OEI rating is requested, the following should be conducted for each engine: Demonstrate 30-second OEI power followed by 2-minute OEI power. After 2 minutes reduce and stabilize power to 30-minute or continuous OEI level. Once the power is stabilized reapply 2 minute OEI power. This should result in 4½ minutes per cycle for a total of 10 minutes at 30-second OEI power and 80 minutes at 2-minute OEI power during the 200-hour endurance test. See figure AC 29.923A-3 for a graphic description of the takeoff run for a two-engine rotorcraft (refer to § 29.923(b)(3)). If either the 2½-minute or 30-second/2-minute OEI ratings are demonstrated, the takeoff run portion of figure AC 29.923A-1 will be longer than 1 hour as shown in figure AC 29.923A-2 or figure AC 29.923A-3, respectively.

2. Apply the rotor brake during the first minute of the 5-minute idle period. Conduct 400 brake applications during the 200-hour endurance test (§ 29.923(j)).

3. During the maximum continuous run, cycle the rotor controls 15 times per hour: (Refer to §§ 29.923(c)(1) - (3)). The cyclic control should be cycled through maximum vertical thrust, maximum forward, maximum left, maximum right, and maximum rearward thrusts. The pedal controls should be cycled through maximum right, neutral, and maximum left positions. Each maximum cyclic and pedal control position should be held for at least 10 seconds. During the remainder of the test, set the yaw control to react to the main rotor torque, and set the flight controls to achieve:

Condition	Portion of Test
max vertical thrust	20 percent
max forward thrust	50 percent
max left thrust	10 percent
max right thrust	10 percent
max rearward thrust	10 percent

Figures AC 29.923A-1, AC 29.923A-2, and AC 29.923A-3 Notes (continued)

4. The 60 percent maximum continuous run is 2 hours (Refer to § 21 29.923(f)), unless either 30-minute OEI or continuous OEI power is requested. In that case, the 60 percent maximum continuous run is 1 hour.

5. A 1-hour malfunction run (if deemed necessary) or the takeoff run is repeated (without OEI portions). Refer to § 29.923(g).

6. The OEI run defined in § 29.923(k) is not required unless an OEI power rating is requested. If a 30-minute OEI power rating is requested, each engine in sequence should be run at the 30-minute OEI condition for 30 minutes. If a continuous OEI power rating is requested, each engine in sequence should be run at the continuous OEI condition for 1 hour. The total OEI run time may exceed the 1 hour shown in figure AC 29.923A-1.

AC 29.923B. § 29.923 (Amendment 29-31) ROTOR DRIVE SYSTEM AND CONTROL MECHANISM TESTS.

a. Explanation. Amendment 29-31 added § 29.923 (p) that defines qualification tests for lubricants used in the rotor drive system and control mechanisms. Section 29.923(p) contains a requirement for a portion of the system qualification tests to be accomplished with specific lubricating oil temperatures and pressures.

b. Procedures. The requirements of § 29.923(p) should be met for all rotor drive system and control mechanism qualification tests. Additionally, these requirements should be met if certification of alternate lubricants for the rotor drive system is requested. Thirty hours of qualification testing is required on the gearbox in which the alternate lubricant(s) is proposed to be used. During this testing, the lubricant temperature is to be measured with the temperature probes that will be used in service. The lubricant temperature should be maintained at the maximum operating temperature established for the gearbox in service. For pressure lubricated systems, the gearbox lubricant pressure should be maintained at the minimum operating pressure that has been established for the gearbox in service. The lubricant pressure is to be measured in the same manner and with the same probes that will be used in service. During the 30 hours of testing required by § 29.923(p), the lubricant temperature and pressure should be applied and measured simultaneously. For one-engine-inoperative ratings, the test time should be extended by one engine failure cycle to include operation at the one-engine-inoperative rating for which approval is requested. Equivalent testing or comparative analysis of the proposed lubricant and the approved lubricant specifications may be used to approve an alternate lubricant.

AC 29.923C. § 29.923 (Amendment 29-34 and 29-40) ROTOR DRIVE SYSTEM AND CONTROL MECHANISM TESTS.

a. Explanation. This paragraph, AC 29.923C, reflects changes made by Amendment 29-34 and 29-40. Amendment 29-34 added § 29.923(b)(3) that defines qualification tests for 30-second/2-minute OEI ratings. This new paragraph also allows for the 30-second/2-minute OEI portion of the endurance test to be accomplished on a representative bench test stand using the drive system components which can be adversely affected by these tests. Amendment 29–40 doubles the endurance test time for 2-minute OEI for each power section.

b. Procedures.

(1) For accomplishment of the endurance test for 30-second/2-minute OEI, § 29.923(b)(3) requires that immediately following one of the 5-minute power on takeoff runs of § 29.923(b)(1) each engine must simulate a power failure and each engine providing power after the failure must apply the maximum torque and maximum speed for use with 30-second OEI power. This power level should be maintained for at least 30-seconds. The 30-second OEI power should then be followed by an application of the maximum torque and maximum speed for 2-minute OEI power for at least 2 minutes. After the 2-minute OEI power application the power should be reduced to and stabilized at 30-minute or continuous OEI, (whichever rating the rotorcraft will be certified with). After the power has been stabilized, the maximum torque and maximum speed for use with 2-minute OEI power should be reapplied for at least 2 minutes. Figure AC 29.923A-1 shows a graphic representation of the ten-hour test cycle with the 30-second/2-minute OEI segment included for each engine presented in figure AC 29.923A-3. This figure shows the OEI test segment being accomplished immediately following a 5 minute takeoff run. The OEI test segment can be accomplished after any of the 5-minute takeoff run segments. Section 29.923(b)(3) also requires that one of the 30-second/2-minute OEI segments for each engine be accomplished from the flight idle condition.

(2) Additionally, due to the damage inflicted on the engines and the ensuing cost caused by operating the engine at these powers, the 30-second/2-minute portion of the endurance test can be accomplished on a bench test rig found to be representative of the rotorcraft. The representative bench test rig should have the ability to generate the torques, speeds, torsional vibration frequency, and engine acceleration rate generated by the actual installation. The power should have the same method/path of application as that used on the rotorcraft. The test rig should be configured with the same components used for conducting the endurance test on the rotorcraft except that the test components not affected by asymmetric power application may not have to be installed (i.e., if a separate combining gearbox is used it may not be necessary to have the main transmission installed on the bench test rig).

(3) When conducting the bench test for 30-second/2-minute OEI, it is not necessary to reaccomplish the takeoff portion of the endurance test. The simulated

power failure and application of 30-second/2-minute OEI power by the remaining power section should be accomplished after the input power has stabilized at takeoff power. The takeoff portion of the endurance test should be accomplished on the rotorcraft.

AC 29.927. § 29.927 (Amendment 29-17) ADDITIONAL TESTS.

a. Section 29.927(a):

(1) Explanation. This paragraph is authority to require any special tests or investigations to establish that the rotor drive system is safe.

(2) Procedures. The certification engineer should review the design of the rotor drive system and its installation and intended operation for features or conditions that may not be adequately qualified in the tests prescribed by this Part. Additional qualification test programs should be developed and accomplished to ensure safe operation of the system. Items of interest would include poorly defined load paths associated with redundant design features, flight deflections of structure, mounting arrangements which may not be properly qualified by ground tests, and special or unusual operating procedures which are anticipated by the applicant.

b. Section 29.927(b):

(1) Explanation. This paragraph prescribes testing to qualify the rotor drive system for the power excursions to be expected with governor-controlled engines wherein the engine power changes automatically to maintain rotorspeed at preselected values. At high collective flight control displacements, the normal rotor speed droop will result in governor-controlled engines automatically accelerating to maximum fuel flow or to any other power, speed, temperature, or torque limiting device, regardless of crew action or artificially established limitations reflected by instrument markings. This high power condition can occur typically during a normal landing when the crew applies high collective to cushion ground contact or, for multiengine rotorcraft, during any flight regime when an engine fails and the corresponding loss of power results in drooping the rotor speed. Special tests are prescribed by this section to provide assurance that the rotor drive system can safely sustain these conditions. The tests of this section should be conducted without intervening disassembly, and all rotor drive system components should be in serviceable condition after the test. It is permissible but not required that these tests be performed on the same specimen of the rotor drive system used to show compliance with § 29.923.

(2) Procedures. These tests should be conducted on a ground-test rotorcraft conformed to the type design configuration similar to that required for endurance testing under § 29.923. Cyclic and collective control may be set to simulate vertical lift and antitorque control set and/or adjusted to react to main rotor torque. Rotation speed should be maximum normal for the test condition; i.e., for the all-engines test under § 29.927(b)(1), use the maximum RPM for takeoff power. For the one-engine-inoperative (OEI) test of § 29.927(b)(2), RPM droop, if any, that would

occur in service, may be allowed. Since the OEI test of § 29.927(b)(2) usually requires the remaining engine(s) to produce power not usually available under normal atmospheric conditions, some supplemental method, such as refrigerating and/or ramming inlet air, or overfueling the engine, may be required. Alternatively, bench testing (transmission test rig) of the rotor drive system (using only the components subject to the higher OEI power, if desired) may be appropriate providing close simulation of the rotor drive system installation environment is achieved. Overtesting, to compensate for inadequacies in the bench test setup may be negotiated with the FAA/AUTHORITY approval office. Note that compliance with § 29.903(b) requires that the remaining engine(s) be capable of continued safe operation under the same conditions as dictated by this test. The engine manufacturer may have already conducted tests adequate to substantiate this requirement. If not, his assistance in testing and the subsequent serviceability finding is imperative.

c. Section 29.927(c):

(1) Explanation. This paragraph prescribes a test which is intended to demonstrate that in the event of a major failure of the lubrication system used on the rotor drive system, no hazardous failure or malfunction will occur in the rotor drive system that will impair the capability of the crew to execute an emergency descent and landing. The lubrication system failure modes of interest usually are limited to failure of external lines, fittings, valves, coolers, etc., of pressure lubricated transmissions and/or gearboxes.

(2) Procedures. Conventionally, a bench test (transmission test rig) is used to demonstrate compliance with this rule. Since this is essentially a test of the capability of the residual oil in the transmission to provide limited lubrication, a critical entry condition for the test would be the critical eligible lubricant preheated to the transmission oil temperature limit. With the transmission operating at maximum normal speed, with lubricant as described above, with nominal cruise torque applied (reacted as appropriate at main mast and tail rotor output quills), and with a vertical load at the mast equal to gross weight of the rotorcraft at 1g, disconnect or cause a leak in an external oil plumbing device. Upon illumination of the low oil pressure warning (required by § 29.1305), reduce input torque to simulate autorotation and continue rotation for 15 minutes. Apply input torque to simulate a minimum power landing for approximately 15 seconds to complete the test. Successful demonstration may involve limited damage to the transmission provided it is determined that the autorotative capabilities of the rotorcraft were not significantly impaired.

d. Section 29.927(d):

(1) Explanation. This test is intended to demonstrate that overspeed conditions which may result from control failure or control misapplication will not incur damage to the rotor drive system. Specific conditions for conducting the test are provided in § 29.927(d)(1), (d)(2), and (d)(3).

(2) Procedures. The test may be conducted on a rotorcraft configured for the endurance tests prescribed by § 29.923. Turbine engines involved in the test may require fuel control rerigging or operation on the manual fuel control system, if available, to achieve test requirements.

NOTE: Some equivalent safety findings have been issued based on limiting the test speed to that permitted by an independent, reliable overspeed trip device, thus avoiding permanent damage to yokes, engines, etc., involved but not subject to evaluation under this rule.

(3) With collective control set for minimum rotor pitch for smooth operation, the cyclic control positioned for vertical lift, and the antitorque control set in flat pitch, add power to achieve 120 percent of maximum continuous speed and hold this condition for 30 seconds. Deceleration and operation between overspeed runs should be as described in the rule. Acceleration and deceleration must be at maximum rates available to the configuration.

e. Section 29.927(e):

(1) Explanation. This paragraph sets forth conditions to be normally employed during the overtorque and overspeed tests of this section and authorizes certain exceptions with criteria for justification.

(2) Procedures. None.

AC 29.927A. § 29.927 (Amendment 29-26) ADDITIONAL TESTS.

a. Explanation. Amendment 29-26 revises and extends the rotor drive system lubrication failure test requirements for Category A rotorcraft in § 29.927(c). Category A rotorcraft should have significant continued flight capability after a failure in order to optimize eventual landing opportunities. Indefinite flight following the lubrication system failure is not expected. The change to the overspeed test requirements in § 29.927(d) removes the arbitrary requirement of 120 percent and substitutes a more realistic limit related to the operating characteristics of the rotorcraft. This change is needed because the existing 120 percent overspeed requirement may be unnecessarily severe for some rotorcraft. An additional change eliminates the requirement for accomplishing the acceleration phase of the overspeed tests within 10 seconds when the maximum acceleration rate of the engine requires more time. This will avoid special engine fuel control modifications for test purposes which are unnecessary and may damage the engine. Section 29.927(f) was added which requires each individual test specified by this section to be conducted without intervening disassembly and, except for the lubrication failure tests of § 29.927(c), requires each part tested to be in a serviceable (return to service) condition at the conclusion of the test. Intervening disassembly is unacceptable since it can invalidate the required serviceability findings. The serviceability requirement is needed to ensure that only test results which are satisfactory may be used to show compliance.

b. Procedures.

(1) Section 29.927(c) prescribes a test which is intended to demonstrate that no hazardous failure or malfunction will occur in the event of a major rotor drive system lubrication failure. The lubrication failure should not impair the ability of the crew to continue safe operation of Category A rotorcraft for at least 30 minutes after perception of the failure by the flight crew. For Category B rotorcraft, safe operation under autorotative conditions should continue for at least 15 minutes. Near the completion of the lubrication failure test, an input torque should be applied for 15 seconds to simulate a minimum power landing following autorotation. Some damage to rotor drive system components is acceptable after completion of the lubrication system testing. The lubrication system failure modes of interest are usually limited to failure of bearings, gears, splines, clutches, etc., of pressure lubricated transmissions and/or gearboxes. A bench test (transmission test rig) is commonly used to demonstrate compliance with this rule. Since this is a test of the capability of the residual oil in the transmission to provide limited lubrication, a critical entry condition for the test should be established. The transmission lubricating oil should be drained while the transmission is operating at maximum normal speed and nominal cruise torque (reacted as appropriate at the main mast and tail rotor output quills). A vertical load should be applied at the mast, equal to the gross weight of the rotorcraft at 1g, and the lubricant should be at the maximum temperature limit. Upon illumination of the low oil pressure warning required by § 29.1305, reduce the input torque for Category A rotorcraft to the minimum torque necessary to sustain flight at the maximum gross weight and the most efficient flight conditions. To complete the test, apply an input torque to the transmission for approximately 25 seconds to simulate an autorotation. The last 10 seconds should be at the torque required for a minimum power landing. A successful demonstration may involve limited damage to the transmission, provided it is determined that the autorotative capabilities of the rotorcraft were not significantly impaired. For Category B rotorcraft, upon illumination of the low oil pressure warning light, reduce the input torque to simulate an autorotation and continue transmission operation for 15 minutes. To complete the test, apply an input torque to the transmission for approximately 10 seconds to simulate a minimum power landing. A successful demonstration may involve limited damage to the transmission provided it is determined that the autorotative capabilities of the rotorcraft were not significantly impaired. If compliance with Category A requirements is demonstrated, Category B requirements will have been met.

(2) Section 29.927(d) provides the requirement to demonstrate that overspeed conditions, which may result from control failure or control misapplication, will not result in damage to the rotor drive system. The overspeed endurance cycle and overspeed conditions to be demonstrated are defined in this section. The test may be conducted on a rotorcraft configured for the endurance tests prescribed by § 29.923. Turbine engines involved in the test may require fuel control re-rigging or operation on a manual fuel control system to achieve test requirements. Fifty overspeed runs of 30 ±3 seconds should be run on the rotor drive system. The overspeed runs should be

alternated with stabilizing runs of from 1 to 5 minutes duration each at 60 to 80 percent of maximum continuous speed.

(i) The maximum speed to be demonstrated during the power on overspeed test is:

(A) The higher of:

(1) The speed to be expected from an engine control device failure; or,

(2) 105 percent of the maximum rotational speed to be expected in service, including transients.

(B) The maximum speed allowed by a speed limiting device if the device is installed independent of the engine controls and is shown to be reliable.

(ii) From the stabilizing run condition, increase power to achieve the maximum speed established from (i) above. Set the collective for minimum blade pitch for smooth operation. The cyclic control should be positioned for vertical lift, and the anti-torque control should be set in flat pitch. Hold this condition for 30 seconds, then decelerate to the stabilizing run condition.

(iii) The acceleration and deceleration described above should be accomplished in 10 seconds or less except where it can be shown that the certified engine acceleration or deceleration rate exceeds 10 seconds. The time required for acceleration and deceleration may not be deducted from the 30 second overspeed period.

NOTE: Some equivalent safety findings have been issued based upon limiting the test speed to that permitted by an independent, reliable overspeed trip device. This has been done to avoid permanent damage to rotors, yokes, engines, etc., which are involved, but not under evaluation by this test.

(3) Paragraph (f) requires that the overtorque, lubrication system failure, and overspeed tests required by §§ 29.927(b), (c), and (d) respectively be conducted without intervening disassembly during the individual test. After each test, a teardown inspection is performed, and except for the components used in the lubrication system failure test, the components are required to be in serviceable (return to service) condition.

AC 29.931. § 29.931 (Amendment 29-12) SHAFTING CRITICAL SPEEDS.

a. Explanation.

(1) At certain speeds, rotating shafts tend to vibrate violently in a transverse direction. These speeds are variously known as “critical speeds,” “whirling speeds,” or

“whipping speeds.” The vibration results from the unbalance of the rotating system and can be shown to reach destructive values with only minimal unbalance. The nature of this phenomena is that as shaft rotational speed increases, residual unbalance in the shaft gives rise to centrifugal forces. These forces cause the shaft to rotate in a bent or bowed configuration with the centrifugal force induced bending loads being balanced by coriolis and elastic forces in the shaft. As shaft rotational speed increases, the centrifugal forces increase to the point at which they exceed the elastic forces in the shaft, and divergence occurs. This point in the speed range is called the critical speed. At shaft speeds above the critical speed, a 180° phase change occurs; the shaft’s mass center moves toward the center of rotation and the amplitude of vibration diminishes with further increases in shaft speed.

(2) The most prominent design option is to operate the shafting subcritical; i.e., below the first critical speed, with adequate margins from critical speed at the maximum allowable speed, including transients. However, another option, that of supercritical shaft operation; i.e., operating above the first or even higher critical speeds with adequate margins between any critical speed for the normal operating speed range. This latter portion requires some form of fixed system damping to permit safe transition through the critical speed range and to avoid excessive nonsyncronous vibrations or instability in the critical speed mode at suboperating frequency.

(3) A review of typical design practices and drive system arrangements discloses several types of shaft support and loading:

(i) Main rotor/mast/transmission assemblies rigidly mounted to the airframe;

(ii) Main rotor/mast/transmission assemblies compliantly mounted to the airframe;

(iii) Main rotor supported through a bearing arrangement by a rigid nonrotating structure with a coaxial torque shaft driving the rotor;

(iv) Cross-shafting, interconnect shafting, tail rotor drive shafting which are generally supported by gearboxes at each end and by hanger bearings at semispan;

(v) Engine to transmission shafting which, for compliant pylons, incorporate flexible or geared coupling, to accommodate the misalignment and chucking; and

(vi) Tail rotor/mast/gearbox supported on the tailboom or near the upper extremity of a vertical fin.

(4) With regard to compliant pylon mountings, recent developments in vibration control have led to rotor isolation wherein the fuselage is isolated from the rotor and

transmission, resulting in improved vibration and system reliability. Rotor isolation systems typically entail the installation of isolation devices at the transmission-airframe interface. The crux of rotor isolation is providing adequate, low-frequency isolation without excessive relative displacement or loss of mechanical stability. Rotor isolation affects shaft critical speeds in the following ways:

(i) First, the transmission mounting configuration, system stiffness, and tuning requirements may result in different fore-and-aft and lateral natural frequencies, imposing additional analytical requirements. For compliant mounting, the response while transitioning through the fundamental or rocking modes is generally controlled by dampers or elastomeric elements.

(ii) Second, the relatively high displacements permitted by the isolation system, depending on configuration, may result in variations in shaft misalignment and length thus adding further complexity to the analytical prediction of critical speeds.

b. <u>Procedures</u>.

(1) <u>Subcritical Shafting Designs</u>. Three basic methods of qualification may be considered, with the required margins relative to the degree of assurance provided:

(i) <u>Analytical</u>.

(A) Simplistic model(s) as shown in figures AC 29.931-1 and AC 29.931-2; 35 percent margin shown above maximum operating speed.

(B) Detailed model, taking into account significant variations in shaft stiffness, mass distribution, cone adapters, support bearing stiffness, support structure; 20 percent margin shown above maximum operating speed.

(ii) <u>Analytical supported by tests</u>. Analysis supported by shake test (rotating or nonrotating) or by bench test, where appropriate adjustments are made for differences between the bench and the aircraft; 15 percent margin shown above maximum operating speed.

(iii) <u>Whirl test on the aircraft</u>.

(A) For all cases, it should be shown that, under maximum permissible unbalance and at the maximum operating speed, the shafting and support structure has acceptable clearance and does not have excessive vibration.

(B) For compliant pylon mountings, damping of the rigid body rocking modes, which are often transitioned during run-up to normal speed (and which are not critical flexing modes), may be verified by analysis, laboratory tests, or ground run-up with the rotor at maximum permissible unbalance. Damping on the order of 5 percent equivalent viscous damping is generally acceptable.

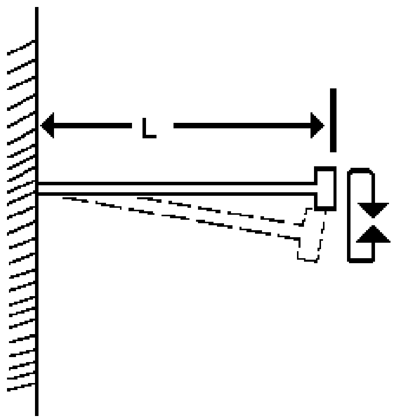
(C) For tail rotor masts, the analysis should include fixed system structural response including tailboom, fixed control surfaces, and vertical fin. The frequency analysis will then contain both fixed system and rotating system modes. An energy analysis can then be used to identify whether the modes are predominantly fixed system or rotating system modes. Systems with up to 35 percent energy in the rotating system have been operated in the field without significant problems. For this type of shafting installations, it is advisable to avoid fixed system modes at multiples of shaft speed, particularly where highly nonisotropic mountings exist.

(2) Supercritical Shafting Design. Another facet occasionally encountered with shafting is the concept of normally operating at speeds above the critical speed, commonly referred to as “supercritical operation.” To function properly, suitable dampers must be installed to enable the shaft to pass safely through the lower critical speed up to the operating speed, and speed controls should be devised so as to avoid any tendency to operate continuously at any critical speed. Accurate balancing of the rotating components will also decrease the energy to be dissipated into the damping device during transition thereby increasing its serviceability and reliability. It should be noted that damper design and locations become more complex as selected operating speed increases through the third or fourth critical frequency. Multiple node points will exist where dampers will not be effective. Production specimen testing at high speed/high torque conditions should include checks for shaft straightness until experience verifies that shaft deflecting is not significant. For system utilizing squeeze film dampers at the support bearings, variations in oil pressure, flow restrictions, and the effects of bearing preload should be evaluated. The effects of shaft and unbalance and the proximity of the damper to bottoming under maximum unbalance should be evaluated.

(3) If the shafting configuration of the rotorcraft includes universal joints or misalignment couplings, a velocity differential will exist across the joint which creates sinusoidal torques and bending moments at both shafts at multiples of the rotation speed. To avoid amplification of these torques and bending moments, the design should preclude coincidence of critical speeds and multiples of normal speeds.

(4) Note that failure considerations required under § 29.901(d) may result in abnormal rotational speed and torque excursions. Resulting encounters with critical speeds should not create hazards.

(5) Order 8110.9, Handbook on Vibration Substantiation and Fatigue Evaluation of Helicopters and Other Power Transmission Systems, also addresses this subject. This document is distributed to section level and above in all Regional Aircraft Certification Offices.



$$W_{cr} = \sqrt{\frac{k}{M + 0.23m}}$$

W_{cr} = first critical speed, RAD/SEC

k = shaft spring rate, LB/IN = $3EI/L^3$

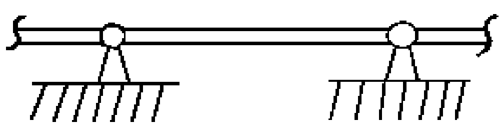
E = modulus of elasticity

I = moment on inertia

M = mass of weight, LB-SEC^2/IN

m = mass of shaft, LB-SEC^2/IN

FIGURE AC 29.931-1. CANTILEVERED SHAFT, FIRST CRITICAL SPEED



$$W_{cr} = a\sqrt{\frac{E\ I}{u\ L^4}}$$

W_{cr} = first critical speed-RAD/SEC

E = Young's modulus

I = inertia of shaft

u = mass per unit length

L = length between supports

a = a numerical constant: for first critical speed, $a = (pi)^2 = 9.87$

The numerical constant (a) for higher critical whirl modes or other shaft support systems may be derived from standard texts on this subject.

FIGURE AC 29.931-2. SHAFT BETWEEN SUPPORT BEARINGS, FIRST CRITICAL SPEED

AC 29.935. § 29.935 SHAFTING JOINTS.

a. Explanation. This rule requires the design of shafting joints to include provisions for lubrication when such lubrication is necessary for operation.

b. Procedures. Review the design of the rotor drive system for universal joints, slip joints (splines), and other shaft couplings. Lubrication access points (Zerk fittings) should be required unless the design incorporates alternate provisions for lubrication acceptable to the FAA/AUTHORITY.

AC 29.939. § 29.939 (Amendment 29-12) TURBINE ENGINE OPERATING CHARACTERISTICS.

a. Explanation. This section requires evaluation of engine operation, engine inlet airflow distortion, and engine/drive system torsional stability. A satisfactory rotorcraft design for all three items should be established by the manufacturer early in his development program since changes in design to satisfy these requirements are typically very expensive and will adversely impact other basic design features. The role of the certification engineer is to assure that the manufacturer's evaluation programs have been thorough and conclusive. The certification engineer should also determine, where applicable, that the FAA/AUTHORITY-approved Engine Installation Manual requirements are met.

b. Procedures.

(1) Turbine engine operation.

(i) Explanation. Smooth, stable operation of turbine engines is essential to safety and control of rotorcraft. This can be adversely affected by rotorcraft maneuvers, turbulence, high altitude, temperature, airspeed, and installation features such as the engine air inlet duct, exhaust duct, and the location with respect to other airframe items which induce or influence air flow through the engine. Powerplant control displacement rate can also be a factor, although most modern engines incorporate internal protection for this aspect. The engine's tolerance to these factors is reflected as the "stall margin" which is established by the engine manufacturer through design and test. However, this stall margin is applicable only to an engine with a specified inlet and exhaust and at specified altitude, temperature, and effective airspeed. Typically, the specified engine inlet duct is a symmetrical bellmouth and the exhaust is a short straight duct of specified diameter and length. The stall margin, even under the above test conditions, usually varies with engine power, acceleration or deceleration, compressor air bleed, and accessory power extraction.

(ii) Procedures. The official flight test plan should include requirements to investigate the engine operating characteristics for stall, surge, flameout, acceleration and deceleration response, and transient response (within approved limits) throughout the operating range of the rotorcraft. This should include maximum airspeed-sideslip combinations, power recoveries, hover with wind from all azimuths and other maneuvers appropriate to the type. Recirculation of exhaust gases during hover can be critical for engine operation. Particular attention should be given to flight/operating conditions which can be judged critical from review of data on engine inlet pressure and temperature distribution patterns and engine stall margin data if available. High altitude has typically been critical for these tests and rearward flight at high altitude has resulted in unacceptable thermal distortions in the inlet due to reingestion. Stall, surge, or flameout which may be hazardous; i.e., causes loss of engine function, loss of control, severe torsional shock through the rotor drive system or otherwise damages the rotorcraft, is unacceptable.

(2) Vibration.

(i) Explanation. Engine airflow patterns are deflected or distorted by the presence of airframe inlet hardware, cowling, fuselage panels, and, to a degree, in almost all flight regimes. Additional items such as airframe installed particle separators, deflectors for snow, ice, or sand protection, and obstructions forward of the engine inlet, such as a hoist kit, could affect the engine air flow patterns. The rotating elements of the engine, particularly the compressor blades, will be subjected to a cyclically varying air flow as these elements move into and out of areas of deflected airflow to the engine. A corresponding aerodynamic load will be imposed on these engine elements. Since this loading is also cyclic, the possibility of critical frequency coupling with an engine component shall be investigated.

(ii) Procedure. Typically, this evaluation would involve installation in the engine inlet of a special multiple probe, total pressure sensing system, and flight testing which largely follows that prescribed for evaluation of engine operating characteristics as described above. Data from these tests can be reduced to create a pressure map at the compressor inlet face which, in conjunction with compressor speeds, may be used to determine the frequencies and relative amplitudes of the cyclic air loading imposed on the engine compressor blades. The engine manufacturer either supplies the sensing probe or specifies its design and performance. Also, the engine manufacturer may evaluate the test results or publish acceptance criteria. A wave analysis may be involved in identifying higher order excitations. Engine exhaust ducts which include bends, noise suppressors, or other obstructions may require an evaluation similar to that discussed above for the engine inlet. The engine manufacturer should be consulted for instructions or approval of this aspect. High performance engines may also require an engine inlet temperature survey. Details of instrumentation and acceptance criteria should be provided by the engine manufacturer. Engines equipped with only centrifugal compressors are less likely to encounter frequency coupling and may not require this investigation. The engine manufacturer's recommendations should be followed in these cases.

(3) Torsional Stability.

(i) Explanation. Governor-controlled engines installed in rotorcraft are subject to a fuel control resonant feedback condition which could be divergent if not properly designed or compensated. This condition occurs when the response frequency of the governor on the engine is coincident with or close to a low order natural torsional frequency of the rotorcraft's rotor drive system. Typically, these frequencies appear in the 3 to 5 cycles per second (CPS) range. The manufacturer usually resolves torsional instability problems by introducing damping into the engine governor/fuel control. Provisions for this change must be supplied by or approved by the engine manufacturer. The final configuration may be a compromise between a lightly damped control, which will allow a positive but slow convergence of drive system torsional oscillations, and a highly damped control which exhibits excessive rotor speed droop or overspeed following rotorcraft collective control displacement.

(ii) Procedures. A ground and flight test program should be devised to evaluate the torsional response of the engine and drive system combination presented by the applicant. Instrumentation to record drive system torsionals should be applied to all major branches of the drive system. Engine parameters such as torque, RPM fuel manifold/nozzle pressure, compressor discharge pressure, and governor lever position should be recorded simultaneously with drive system parameters. The test program should include ground tie-down operation and flight operation across a range of engine power and rotor speeds while injecting control inputs as close to the first order drive system natural frequency as possible. Mechanical methods of making these inputs are not usually necessary if the desired frequency is in the 3 to 5 CPS range and the instrumentation readout confirms that the drive system was actually excited torsionally at its natural frequency. Control inputs should include collective, antitorque, and throttle. Also, cyclic inputs may be important on tandem rotor rotorcraft. The acceptance criteria may be dependent on several items. Among these are rotor and drive system fatigue loading, engine power response characteristics, limitations established by the engine manufacturer, etc. The acceptance criteria are usually stated as a percent damping (minimum). Typically, 1 percent of critical equivalent viscous damping (or greater) is acceptable. In effect, this means that the free vibration response to a control input damps to ½ amplitude in 11 cycles or less.

SUBPART E - POWERPLANT

FUEL SYSTEM

AC 29.951. § 29.951 (Amendment 29-12) FUEL SYSTEM - GENERAL.

a. Explanation.

(1) The term "fuel system" means a system which includes all components required to deliver fuel from the tank(s) to the engine(s). This includes, but is not limited to, all components provided to contain, convey, drain, filter, shutoff, pump, jettison, meter, and distribute fuel to the engines.

(2) Paragraph (a) of this section is a general statement of the performance requirements for fuel systems and constitutes authority to require fuel systems to be adequate notwithstanding compliance with detail requirements listed in §§ 29.953 through 29.999 of this subpart.

(3) Paragraph (b) of this section requires fuel systems to be designed so that air will not enter the system under any operating conditions by either arranging the system so that no fuel pump can draw fuel from more than one tank or by other acceptable means.

(4) Paragraph (c) of this section sets forth a fuel system performance requirement intended to ensure that ice to be expected in fuel when operating in cold weather will not prevent the fuel system from supplying adequate fuel to the engines. Although fuel system filters and strainers are the items in the fuel system most susceptible to clogging from ice particles in the fuel, this paragraph requires that the entire fuel system be shown to be capable of delivering fuel, initially contaminated with ice, to the engine(s).

b. Procedures.

(1) For paragraph (a), the applicant should show compliance with the fuel system requirements of this subpart, except that if unusual fuel system arrangements or requirements exist which are not adequately addressed by these subparts, this paragraph may be used as authority to require special tests, analysis, or system performance needed for proper engine functioning.

(2) For paragraph (b), review the fuel system design with special attention to fuel tank selector valves, crossfeed systems, and multiple tank outlet arrangements to ensure that no allowable fuel system configuration will permit air to enter the system. For questionable situations, the applicant should conduct ground or flight tests, as necessary, to verify compliance with this section.

(3) Paragraph (c) provides for sustained satisfactory operation of the fuel system with cold fuel initially contaminated with water. Since ice in the fuel system is not considered to be an emergency condition but, rather, is an expected service encounter, compliance would not involve the imposition of special rotorcraft limitations. Flight manual instructions such as land as soon as practicable, reduce altitude to some value less than otherwise permitted, reduce power, turn on boost pumps, etc., are not appropriate in demonstrating compliance. Some methods of fuel system ice protection which have been used to show compliance follow.

(i) Fuel heater. Usually these devices are fuel-to-engine oil heat exchangers and are normally located to protect the fuel filter from blockage by ice in the fuel. The adequacy of these devices should be established. Usually this involves generation of a heat balance between heat gained by fuel and heat lost by oil using performance data provided by the manufacturers of the fuel-oil heater, the oil cooler, the heat rejected by the engine to the oil, etc. A minimum oil temperature associated with the adequacy of the fuel heater may need to be established, marked on the oil temperature gauge, and verified to be maintained during critical flight conditions. Other unprotected parts of the fuel system remain to be evaluated and substantiated for compliance with this requirement.

(ii) Oversized fuel filter. This method may only substantiate the fuel filter and, as with the fuel heater method, is incomplete without evaluation of the remainder of the fuel system. An icing test of the filter should be accomplished. Fuel preparation procedures and method of testing should follow the applicable portion of SAE Aerospace Recommended Practice (ARP) No. 1401. A satisfactory configuration is achieved when a filter is demonstrated to have the capacity to continue to provide the filtration function, without bypassing, when subjected to fuel contaminated by ice to the degree required by this rule. Usually, a delta pressure caution signal for the filter is needed to alert the flightcrew that progressive filter blockage is in progress. The caution device setting should be established by test which demonstrates that after illumination of the caution signal sufficient filter capacity exists to enable completion of the flight. Fuel pressure should not fall below established limits because of ice accumulation on the filter.

(iii) Anti-ice additives. This method utilizes the properties of ethylene glycol to reduce the freezing temperature of water in the fuel. It has the advantage over other methods of protecting all components in the fuel system from ice blockage. Compliance with the rule by this method involves the following.

(A) Eligible additives. PFA-55MB (Phillips Petroleum Co.) and additives per specification MIL-I-27868, Revision D, or earlier. Later versions of this specification do not require glycerin, which may be needed to protect fuel tank coatings.

(B) Compatibility. Both engine fuel system and aircraft fuel system should be verified to be chemically compatible with the additive at the maximum concentration to be expected in the fuel system. Usually, information on eligible system materials can be obtained from the engine manufacturer for the engine fuel system and from the additive manufacturer for aircraft fuel system materials.

(C) Adding or blending the additive to the fuel. These additives do not mix well with the fuel and indiscriminate dumping of additive into the tank will not only fail to protect the system from ice accumulation but likely will damage nonmetallic components in the system. Some fuels may have additive premixed in the fuel. If other fuels are to be eligible, a method for blending additive into the fuel during refueling must be devised and demonstrated to be effective.

(D) Placards should be added near the fuel filler opening to note that fuel must contain the approved anti-ice additive within the minimum and maximum allowed concentration.

(E) The FAA/AUTHORITY-approved flight manual should contain necessary information to attain satisfactory blending of the additive and procedures to allow the operator to check the blend in the fuel tank.

(iv) Fuel system protection (other than filters). If the fuel heater method or oversize filter method (paragraphs b(3)(i) and b(3)(ii)) is proposed, the remainder of the fuel system should be shown to be free from obstruction by fuel ice. This may be shown by testing the system with ice contaminated fuel (prepared as suggested for filter tests) or, in many cases, by selecting fuel system components which by test or by previous experience are known to be free of ice collection tendencies. Tank outlet screens (or tank-mounted pump inlet screens) may be the significant fuel system feature for further evaluation. In some instances, fuel turbulence due to pump motions may be sufficient to keep the screen clear of ice. In other instances, small screen bypass openings (approximately one-fourth inch in diameter) located outside the predominant fuel flow path have been found satisfactory.

NOTE: Advisory Circular (AC) 20-29 contains information regarding compliance with the fuel ice protection requirements of Part 25. The information in this AC is largely valid except for references to the quantity of water to be expected in fuel and the amount of additive required to ensure freedom from fuel ice hazards.

AC 29.952. § 29.952 (Amendment 29-35) FUEL SYSTEM CRASH RESISTANCE.

a. Explanation.

(1) Section 29.952 provides safety standards that minimize postcrash fire (PCF) in a survivable impact. The rule contains comprehensive crash resistant fuel system (CRFS) design and test criteria that significantly minimize fuel leaks, creation of potential ignition sources, and the occurrence of PCF. Section 29.952 accomplishes this for survivable impacts by-

(i) Providing comprehensive criteria to minimize fuel leaks and potential ignition sources;

(ii) Requiring increased crash load factors for fuel cells in and behind occupied areas to ensure the static, ultimate strength necessary for impact energy absorption, structural integrity, fuel containment, and occupant safety;

(iii) Maintaining the load factors of § 29.561 for fuel cells in other areas (particularly underfloor cells) to ensure leak-tight fuel cell deformation in energy absorbing underfloor structure without unduly crushing or penetrating the occupiable volume; and

(iv) Requiring a 50 ft. dynamic vertical impact (drop) test to measure fuel tank structural and fuel containment integrity.

(2) Section 29.952 applies to all fuel systems (including auxiliary propulsion unit (APU) systems).

(3) Some similarities exist among the fire protection requirements of §§ 29.863, 29.1337(a)(2), and 29.952. The requirements in each standard are not mutually exclusive. Overlapping requirements should be certified simultaneously.

(4) The use of bladders is not mandated as this would unduly dictate design. However, in the majority of cases, their use is necessary to meet the test requirements of § 29.952. If a design does not use bladders, the application should be treated as a new and unusual design feature that should be thoroughly coordinated with the Airworthiness Authority for technical policy to insure adequate safety. Experience has shown that bladders with wall thicknesses from 0.03 to 0.018 inches typically meet the § 29.952 test requirements.

b. Related Material. Documents shown below may be obtained from The Naval Publications and Forms Center, 5801 Tabor Avenue, Philadelphia, Pennsylvania 19120-5094, ATTN: Customer Service (NPODS).

(1) Military Specification, MIL-T-27422B, Amendment 1, April 13, 1971, Tank, Fuel, Crash-resistant Aircraft.

(2) Military Standard, MIL-STD-1290 (AV), January 25, 1974, Light Fixed and Rotary Wing Aircraft Crashworthiness.

(3) Military Standard, MIL-H-83796, August 1, 1974, Hose Assembly, Rubber, Lightweight, Medium Pressure, General Specification for.

(4) Military Specification, MIL-V-27393 (USAF), July 12, 1960, Valve, Safety, Fuel Cell Fitting, Crash Resistant, General Specification, for.

(5) Military Specification, MIL-H-25579 (USAF).

(6) Military Specification, MIL-H-38360.

(7) U.S. Army Publication USARTL-TR-79-22E, “Aircraft Crash Survival Design Guide, Volume V---Aircraft Postcrash Survival”, dated January 1989.

NOTE: Section 4, “Postcrash Fire Protection” of Volume V of the Design Guide is the modern update to MIL-STD-1290. Section 4 contains a comprehensive design guide for military CRFS designs that may be useful for civil CRFS designs.

c. Conceptual Definitions.

(1) Survivable Impact. An impact (crash) where human tolerance acceleration limits are not exceeded in any of the principal rotorcraft axes, where the structure and structural volume surrounding occupants are sufficiently intact during and after impact to constitute a livable volume and permit survival, and where an item of mass does not become unrestrained and create an occupant hazard. “Livable volume” relates to the ability of an airframe to maintain a protective shell around occupants during a crash and to minimize threats, such as accelerations, applied to the occupiable portion of the aircraft during otherwise survivable impacts. In lieu of a more rational, approved criteria, the load factors of § 29.952(b)(1) constitute the structural human survivability accelerations limits.

(2) Postcrash Fire (PCF). A fire occurring immediately after and as a direct result of an impact. The fire is either the result of fuel released from a leaking fuel system reaching an existing or a crash-induced ignition source, a crash-induced ignition source internal to an undamaged or damaged fuel system, or a combination. PCF's have an intensity range from the minimum of a small local flame to the maximum of an instantaneous massive fire or fireball (explosion).

(3) Fuel Tank or Cell. A reservoir that contains fuel and may consist of a hard shell (of a composite, metal, or hybrid construction) with either a laced-in, snapped in, or otherwise attached semirigid or flexible rubber matrix bladder (or liner), spray-on bladder, or no bladder. The hard shell may be either the airframe (integral tank) or a separate rigid tank attached to the airframe. The device has inlets and outlets for fuel transfer and internal pressure control.

(4) Ignition Source. An ignition source that when wet with fuel or in contact with fuel vapor would cause a PCF.

(5) Major Fuel System Component. A fuel system part with enough mass, installation location hazard or a combination to be structurally considered in a crash. Structural consideration is required when crash-induced relative motion can occur between the part and its surrounding structure from inertial impact forces, airframe deformation forces, or for other reasons.

(6) Drip Fence. A physical barrier that interrupts liquid flow on the underside of a surface, such as a fuel cell, and allows it to drip nonhazardously to an external drain.

(7) Flow Diverter. A physical barrier that interrupts or diverts the flow of a liquid.

(8) Frangible Attachment or Fitting. An attachment or fitting containing a part that is designed and constructed to fail at a predetermined location and load.

(9) Deformable Attachment or Fitting. An attachment or fitting containing a part that is designed and constructed to deform at a predetermined location and load to a predetermined final configuration.

(10) Self-Sealing Breakaway Fuel Fitting. A fuel-carrying in-line, line-to-firewall, bulkhead or line-to-tank connection that breaks in half and self-seals when subjected to forces greater than or equal to the unit's design breakaway force. Each half self-seals using a spring-loaded valve (e.g., trap door or equivalent means) that is normally open but is released and closed upon fitting separation. Fitting breakaway force is typically controlled by a frangible metal ring (or series of circumferential tabs) that connects the two fitting halves. Normal, fuel-tight integrity is maintained by "O" rings held under pressure by the rigid, frangible connecting ring (or tabs). When broken open, a small amount of fuel (usually less than 8 ounces) is released. This is the fuel trapped in the coupling space between the two spring-loaded valves. Once failed each coupling half may leak slightly. Typically, this leak rate should be less than 5 drops per minute per coupling half.

(11) Crash Resistant Flexible Fuel Cell Bladder. Flexible, rubberized material, usually with fibers (i.e., rubber "resin" and natural or synthetic fiber) in both the 0° (warp) and 90° (fill) directions that is used as a liner in a rigid shell or integral tank. The material acts as a membrane because, when unsupported, it can only carry pure tension loads. Therefore, it must be uniformly supported by rigid structure (reference § 29.967) so that the liner carries only compressive fluid loads and the surrounding shell structure carries the fluid-induced shear, tension, and bending loads transmitted through the liner or bladder. The material is usually secured (e.g., laced, snapped, etc.) into its surrounding structure at key locations to maintain its intended conformal shape. In many designs, lightweight spacers, such as structural foam, are used between the liner and the airframe to maintain the liners intended conformal shape and to transmit fluid loads to the airframe. The material is either qualified under TSO-C80, "Flexible Fuel and Oil Cell Material," or qualified during certification. Sections 29.952 and 29.963(b) have increased the minimum puncture resistance qualification requirement for liner material (See TSO-C80, paragraph 16.0) from 15 to 370 pounds.

(12) Crash Resistant Fuel System (CRFS). A fuel system designed and approved in accordance with § 29.952 that either prevents a PCF or delays the start of a severe PCF long enough to allow escape.

(13) As Far as Practicable. "As Far as Practicable" means that within the major constraints of the applicant's design (e.g., aerodynamic shape, space, volume, major structural relocation, etc.), this standard's criteria should be met. The level of practicability is much higher in a new design project than in a modification project. The engineering decisions, evaluations, and trade studies that determine the maximum level of practicability should be documented and approved.

(14) Fireproof. Defined in § 1.1, "General Definitions" and in AC 20-135, "Powerplant Installation and Propulsion System Component Fire Protection Test Methods, Standards and Criteria" dated February 6, 1990.

d. Procedures.

(1) Section 29.952 should be applied to all fuel system installations. Any major design change should be reevaluated for compliance with the CRFS requirements. It should be noted that most standard materials and processes are acceptable for crash resistant fuel system construction; however, magnesium, magnesium alloys, and cadmium plated parts (when exposed to fuel) are not recommended, because of their inherent ability to create or contribute to a post crash fire. Section 29.952(a) requires each tank, or the most critical tank (if clearly identified by rational analysis) to be drop tested. The tank is filled 80 percent with water and the remaining 20 percent is filled with air (or, in the case of a flexible fuel cell, the air may be evacuated by hand and the cell resealed). The tank openings, except for the vents, are closed with plugs (or other suitable means) so that they remain watertight. The vents are left open to simulate natural venting. Otherwise, the tank is flight configured. The test tanks are installed in their surrounding structure and dropped from a height of 50 feet on a nondeformable surface (e.g., concrete or equivalent). To be considered a valid test, the tank must impact horizontally ±10°. The 50-foot distance is measured between the nondeformable surface and the bottom of the tank. The ±10° attitude requirement can be ensured by using lightweight cord or a light sling to balance the tank assembly horizontally prior to being dropped. MIL-T-27422B shows a typical test setup. Tank attitude at impact should be verified by photography or equivalent means. The nondeformable floor surface should be covered by a thin plastic sheet so that any leakage is readily detected. The tank water should be tinted with dye to make leakage and seepage sources easy to identify. The tank (except for the vent openings) should be wrapped in light plastic sheet to ensure that minor leakage or seepage (and its source) is detected. Minor spillage through the open vents during the drop test is allowed. The dye should not significantly affect the water's viscosity or other physical properties that may reduce or eliminate any leakage from the drop test. The nondeforming drop test surface should be carefully reviewed. Concrete is acceptable. A fixed and uniformly supported steel plate (loaded only in uniform compression without any springback) is acceptable. Floors or floor coverings such as dirt, clay, wood, or

sand are not acceptable. Selection of the critical fuel tank is important. Factors such as size, fuel cell design and construction, and material(s) should be accounted for when selecting the critical tank. The applicant may elect to drop only a bare fuel cell, not a surrounding structural airframe segment with a fuel cell installed. If so, the applicant must show that puncture hazards to the fuel cell have been eliminated.

(i) If the applicant elects to perform the drop test with surrounding aircraft structure, the cell should be enclosed in enough surrounding structure (production or simulated) so that the airframe/fuel tank interaction during the 50-foot drop is realistically evaluated. This allows the fuel-tight integrity of the "as installed" fuel cell to be evaluated and may provide protection in some designs due to the energy absorption of the surrounding airframe when crushed by impact. This provides realistic testing of fuel cell rupture points caused by installation design features, projections, excessive deformation and local tearout of fittings, joints, or lacings. The amount of actual (or simulated) structure included in the test requires engineering evaluation, risk assessment, and detailed analysis and may require subassembly (e.g., joint) tests for proper determination. Typically, the structure surrounding and extending 1 foot forward and aft of the fuel cell is adequate. This structure has a high probability of causing crash-induced fuel cell leakage. Each application should be examined individually to include all potential structural hazards. If the surrounding structure is clearly shown not to be a contributing hazard for the drop test, and if the applicant elects to do so, the fuel cell may be conservatively dropped alone. This determination should be carefully made by a detailed engineering evaluation. The evaluation should use standard, finite element-based programs (e.g., 'KRASH", NASTRAN, etc.) or similar programs submitted during certification, subassembly or component tests. Elimination of the surrounding structure for the drop test configuration is not trivial. If elimination is applied for, the data should clearly and conclusively show that the surrounding structure is not an impact hazard. In any case, the drop height is a constant 50 feet. The work that determines the test article configuration should be summarized, documented, and approved.

(ii) If the drop test is used to show partial compliance with the underfloor fuel cell load factors of § 29.952(b)(3), test plans should be approved. Minor spillage from the open vents is allowed. Full compliance to these load factors should be shown by static analysis and/or tests. The intent is to provide a fuel cell that is fuel tight and does not unduly crush the occupiable volume or overly stiffen energy absorbing underfloor structure under vertical impact.

(iii) Immediately after the drop test, the tank should be placed in the same axial orientation from which it was dropped and visually examined for leakage. Minor spillage from the open vents is allowed. After 15 minutes, the tank should be reexamined and any new leakage or seepage sources noted and recorded. Any evidence of fluid on the plastic floor cover or tank wrapping sheet should be noted and recorded. Any fluid leakage or seepage constitutes a test failure. This procedure should be repeated immediately with the tank inverted and the vents plugged. The inversion procedure will identify any leak sources on the upper surfaces.

(2) Section 29.952(b) provides three sets of static load factors for design and static analysis of fuel tanks, other fuel system components of significant mass and their installations. “Installation” is structurally defined as the fuel cell’s attachment to the airframe and any additional local (point design) airframe structure affected significantly by fuel cell crash loads (i.e., that would fail or deform to the extent that a fuel spill or a ballistic hazard would occur in a survivable impact). Section 29.952(d) significantly limits the amount of local airframe structure to be considered. The provision of load factors by zone ensures the fuel-tight integrity necessary to minimize PCF in a survivable impact. Unless explicitly shown by both analysis and test that the probability of fuel leakage in a survivable impact is 1×10^{-9} or less, each tank and its installation must be designed and analyzed to one set of these load factors. Also, as stated and explained in the advisory material for § 29.561, the load factors specified by § 29.561(d) are for the airframe structure surrounding the fuel cell only. The fuel cells themselves (and any fuel system components of significant mass in the underfloor area) and their attachments to the surrounding airframe structure are subject to the load factors of § 29.952(b)(3).

(i) Section 29.952(b)(1) provides load factors for the design and static analysis of fuel cells and their attachments inside the cabin volume. These load factors are provided to prevent crash-induced fuel cell ballistics hazards to and fuel spills (that may cause a PCF) directly on occupants from local structural failures in a survivable impact.

(ii) Section 29.952(b)(2) provides load factors for design and static analysis of fuel cells and their attachments located above or behind the cabin volume. These load factors are provided to prevent injury or death from a fuel cell behind or above the occupied volume that is loosened by impact and to prevent fuel spills (which may cause a PCF) in a survivable impact.

(iii) Section 29.952(b)(3) provides load factors identical to those of § 29.561 for design and static analysis of fuel cells and attachments located in areas other than inside, behind, or above the cabin volume. Since many fuel cells are located under the cabin floor, these load factors provide fuel-tight structural protection in a survivable impact.

(iv) For some crash resistant semi-rigid bladder and flexible liner fuel cell installations, the 50-foot drop test (reference § 29.952(a)) can (with some additional rational analysis) simultaneously satisfy both the drop test requirement and the vertical down load factor ($-N_Z$) requirement of § 29.952(b)(3) for the fuel cell itself and its installation. This approach reduces the certification burden.

(v) For applicants that seek to substantiate the $-N_Z$ load factor requirement of § 29.952(b)(3) using the 50-foot drop test, additional substantiation is required for § 29.952(b)(3) (as is currently practiced) for the fuel cell under the loading of the remaining three load factors and the remaining rotorcraft structure under the loading of all four load factors. In some cases, substantiation of the remaining three load factors can be further simplified by a successful drop test if the fuel cell is symmetric (i.e., structurally equivalent in all four directions).

(3) Section 29.952(c) requires self-sealing breakaway fuel fittings at all fuel tank-to-line connections, tank-to-tank interconnects, and other points (e.g., fuel lines penetrating firewalls or bulkheads) where a reasonable probability (as determined by engineering evaluation, service history, analysis, test or a combination) of impact-induced hazardous relative motion exists that may cause fuel leakage to an ignition source and create a PCF during a survivable impact. In some coupling installations (such as fuel line-to-fuel tank connections), the tank coupling half should be sufficiently recessed into the tank or otherwise protected so that hazardous relative motion (of the fuel cell relative to its surroundings) following an impact-induced coupling failure does not cause a tearout or deformation of the tank half of the separated coupling that would release fuel. The only exceptions are either-

(i) Installations that use equivalent devices such as extensible lines (hoses with enough slack or stretch to absorb relative motion without leakage) or motion absorbing fittings (rotational or linearly extensible joints); or

(ii) Installations that conclusively show by a combination of experience, tests, and analysis to have a probability of fuel loss to an ignition source in a survivable crash of 1×10^{-9} or less.

(4) Section 29.952(c)(1) specifies the basic design features required for self-sealing breakaway couplings.

(5) Section 29.952(c)(1)(i) defines the design load (strength) conditions necessary to separate a breakaway coupling. These loads should be determined from analysis and/or test, reference paragraph d(6). The minimum ultimate failure load (strength) is the load that fails the weakest component in a fluid-carrying line based on that component's ultimate strength. This load comes from local deformation between the coupling and its surrounding structure during a worst-case survivable impact. A failure test of three specimens of the weakest component in each line that contains a coupling should be conducted in the critical loading mode. (If a single critical loading mode cannot be clearly identified, each of the three most critical loading modes should be tested.) The three specimen test results should be averaged. The average value is then used to size the breakaway fuel coupling. [For standard specification (i.e., "off the shelf") hardware, equivalent testing may have already been accomplished and, if no other mitigating circumstances in the design and installation exist, need not be repeated.] To assure separation of the coupling prior to fuel line failure and to prevent inadvertent actuation, the design load that separates the coupling should be between 25 and 50 percent of the minimum ultimate failure load (strength) of the line's weakest component. The critical loads should be compared to the normal service loads calculated and measured at the coupling location to insure unintended service failures do not occur. Typically this criterion is readily satisfied by the natural design because

working loads are much less than crash-induced loads. A separation load less than 300 pounds should not be used regardless of the line size. The minimum 300-pound load is necessary to prevent ground maintenance failures. A fatigue analysis and/or test (reference paragraph d(10)) should be performed to ensure the installation is either a safe-life design or has a conservative, mandatory replacement time. The simplified method of section 9(a) of AC 20-95 may normally be used because of the low ratio of working-load-to-crash-induced failure load. However, since fatigue failures have occurred in service, all fatigue sources (especially high-cycle vibratory sources) should be evaluated. Fracture critical materials should be avoided, and damage tolerant materials utilized. Also, if airframe deformation due to flight loads is significant, its effect on the couplings should be checked to ensure that static or low-cycle fatigue failures do not occur prior to the part's intended retirement life. Large flight load deformations are not usually present in rotorcraft.

(6) Section 29.952(c)(1)(ii) requires a self-sealing breakaway coupling to separate when the minimum breakaway load (reference paragraph d(5) and § 29.952(c)(1)(i)) is met or exceeded in a survivable impact. The loading modes (each of which produces a breakaway load) are determined by analyzing and/or testing the surrounding structure to determine the probable impact forces and directions. The modes usually occurring are tension, bending, shear, compression, or a combination (reference figure AC 29.952–1). The coupling should be designed and tested to separate at the lowest ultimate impact load (lowest critical mode) as long as the minimum working load criterion of § 29.952(c)(1)(i) is also satisfied. Each breakaway coupling design should be tested in accordance with the following (reference MIL-STD-1290) or equivalent procedures. It should be noted that the ratio of the ultimate failure load of the weakest component in the fuel line and the normal service load (i.e., the peak load or approved clipped peak load experienced during a typical flight) of that component should be as high as possible and still meet the other load criteria of this section. Typically, this ratio should not be less than 5.

(i) Static Tests. Each breakaway coupling design should be subjected to tension and shear loads to verify and establish the design load required for separation, nature of separation, leakage during valve actuation, general valve functioning, and leakage following valve actuation. The rate of load application should not be greater than 20 inches per minute. Tests to be used where applicable are shown in figure AC 29.952-1.

(ii) Dynamic Tests. Each breakaway coupling design should be proof-tested under dynamic loading conditions. The couplings should be tested in the three most likely anticipated modes of separation as defined in paragraph d(5). The test configurations should be similar to those shown in figure AC 29.952-1. The load should be applied in less than 0.005 second, and the velocity change experienced by the loading jig should be 36 ±3 feet per second.

(7) Section 29.952(c)(1)(iii) requires that breakaway couplings be visually inspectable to determine that the coupling is locked together (fuel-tight) and remains

open during normal operations. Visual means (such as, an axial misalignment between the two coupling halves, a designed-in visual indicator, a combination or other acceptable criteria) should be considered and specified in the maintenance manual rejection criteria for operational inspections. Inspectability and phased inspection requirements should be evaluated. Special inspections after severe maneuvers or hard landings should be required.

(8) Section 29.952(c)(1)(iv) requires breakaway couplings to have design provisions that prevent uncoupling or unintended closing by operational shocks, vibrations, or accelerations. These provisions depend on both the coupling's design and installation location. The structural environment should be defined, analyzed, and compared with coupling specifications and certification data so that inadvertent decoupling or closing does not occur. A phased inspection requirement should be considered.

(9) Section 29.952(c)(1)(v) requires a coupling design to not release more than its entrapped fuel quantity when the coupling has separated and each end is sealed off. The entrapped fuel is determined by the coupling design and is essentially the fuel trapped between the seals when separation occurs (See breakaway coupling definition). This is usually less than 8 ounces of fuel per coupling. Most coupling designs will leak slightly after separation. This is acceptable but the leak rate should be 5 drops per minute, or less, per coupling half. Specifications defining the entrapped volume of fuel should be approved. If the coupling is not approved or manufactured to an acceptable military or civil specification, the qualification testing of d(6) should be conducted.

(10) Section 29.952(c)(2) requires that each breakaway coupling or equivalent device either in a single fuel feed line or a complex fuel feed system (e.g. a multiple feed line or multitank cross feed system) be designed, tested, installed, inspected, maintained, or a combination, so that the probability of inadvertent fuel shutoff in flight is 1×10^{-5}, or less, as required by § 29.955(a). This should be determined by reliability and failure analysis, other analysis, tests, or a combination and should be documented and approved. Continued airworthiness should be ensured by phased inspections, specific component replacement schedules, or a combination. This section also requires each coupling or equivalent device to meet the fatigue requirements of § 29.571 to prevent leakage. (See the fatigue discussion in paragraph d(5).) The typical method of compliance with § 29.571 used for rotor system parts may not be necessary to meet § 29.952(c)(2). An S-N curve may not need to be generated using full-scale specimen fatigue tests if the conservative method of Section 9(a) of AC 20-95, "Fatigue Evaluation of Rotorcraft Structure" can be applied successfully.

(11) Section 29.952(c)(3) requires that an equivalent device, used instead of a breakaway coupling, not produce a load, during or after a survivable impact, on the fuel line to which it attaches greater than 25-50 percent of the ultimate load (strength) of the line's weakest component. This minimizes crash-induced fuel spills that may cause a PCF. The ultimate strength of the weakest component should be determined by

analysis and/or tests. At least three specimens of the component should be tested to failure in the critical loading mode and the results averaged. [For standard specification (i.e., "off the shelf") hardware, equivalent testing may have already been accomplished and, if no other mitigating circumstances in the design and installation exist, need not be repeated.] The average value is then used to size the equivalent device. Each equivalent device must meet the fatigue requirements of § 29.571 to prevent fatigue-induced leakage. Equivalent devices should be statically and dynamically tested in an identical manner (where feasible) to breakaway couplings (reference paragraph d(6)). All fuel hoses and hose assemblies (whether or not they are used in lieu of breakaway fittings) should meet the following (reference MIL-STD-1290) or equivalent requirements. Any stretchable hoses used as equivalent devices should be able to elongate a minimum of 20 percent without leaking fuel. All other hoses used as equivalent devices should have a minimum of 20-30 percent slack. It should be noted that the ratio of the ultimate failure load of the weakest component in the fuel line and the normal service load (i.e., the peak or approved clipped peak load experienced during a typical flight) of that component should be as high as possible and still meet the other load criteria of this section. Typically, this ratio should not be less than 5.

(i) All hose assemblies should meet or exceed the cut resistance, tensile strength, and hose-fitting pullout strength criteria of MIL-H-25579 (USAF), MIL-H-38360, or equivalent standards.

(ii) Hoses should neither pull out of their end fittings nor should the end fittings break at less than the minimum loads shown in figure AC 29.952-3 when the assemblies are tested as described in d(11)(iii) below. In addition to the strength requirements, the hose assemblies should be capable of elongating to a minimum of 20 to 30 percent by stretch, slack, or a combination without fluid spillage.

(iii) Hose assemblies should be subjected to pure tension loads and to loads applied at a 90° angle to the longitudinal axis of the end fitting, as shown in figure AC 29.952-2. Loads should be applied at a constant rate not exceeding 20 inches per minute.

(12) Section 29.952(d) requires frangible or deformable structural attachments to be used to install fuel tanks and other major system components to each other and to the airframe when crash-induced hazardous relative motion could cause local rupture and tearout of the component, spill fuel to an ignition source, and create a PCF. If it can be conclusively determined that the probability of fuel spillage is 1×10^{-9} or less, no further action is required. Typically, frangible designs are much easier to certify than deformable designs because the scatter in failure loads is much less. Also, some standard frangible military hardware (e.g., frangible bolts) is readily available. This is not so for deformable designs. Each frangible or deformable structural attachment and its installation should be reviewed to insure that, after an impact failure (i.e., separation or deformation), it does not become a puncture or tear-out hazard and cause fuel spillage.

(13) Section 29.952(d)(1) defines the impact design load conditions necessary to deform a deformable attachment or to separate a frangible attachment. These loads should be determined from analysis and/or test (reference paragraph d(14)), and verified during certification. All impact loading modes (tension, bending, compression, shear, and a combination) should be analyzed and the minimum critical frangible or deformable design load determined, based on the ultimate strength of the attachment's weakest component. The critical load should be compared to the normal service loads calculated and measured at the attachment's location to insure unintended service failures do not occur. (Normally, this criterion is readily satisfied because working loads are much less than impact loads.) A fatigue check should be conducted to ensure that the attachments meet the requirements of § 29.571. Typically, this can be accomplished using the simplified method of Section 9(a) of AC 20-95 because of the low ratio of working-load-to-crash-induced failure load. However, because of service history, all fatigue sources (especially high cycle vibratory sources) should be reviewed. The standard method of compliance with § 29.571 used for rotor system parts may not be necessary to meet § 29.952(d)(3). An S-N curve may not need to be generated using full-scale specimen fatigue tests, if the conservative method of Section 9(a) of AC 20-95 can be applied successfully. Fracture critical materials should be avoided and ductile, damage tolerant materials utilized. Phased inspections to ensure continued airworthiness should be considered. Special inspections after severe maneuvers or hard landings should be required. A breakaway or deformation load less than 300 pounds (based on maintenance considerations) is not permitted. If airframe deformation due to flight loads is significant, its effect should be checked to ensure that a static failure or low cycle fatigue failure does not occur. Large flight load deflections are not usually present in rotorcraft.

(14) Section 29.952(d)(2) requires a frangible or locally deformable attachment to function when the minimum breakaway or deformation load (reference § 29.952(d)(1)) is met or exceeded in a survivable impact. The minimum breakaway or deformation load is the load that either breaks or deforms each of the frangible or deformable attachment(s) of each fuel cell, fuel line, or other critical fuel system component to the airframe. Each breakaway/deformation load must be between 25 percent to 50 percent of the load which would cause failure (i.e., impact induced tearout and subsequent fuel leakage) of the attachment to fuel cell, fuel line, or other critical component interface. This is necessary in some installations to prevent tearout of the structural attachment from the fuel cell component to which it is attached and the resultant fuel leakage in a survivable impact. The primary loading modes (each of which will produce a breakaway or deformation load) must all be considered to determine the minimum load. This is done by analyzing the surrounding structure (reference paragraph d(13)) to determine the three most probable impact failure forces and their directions. The attachment should then be tested to insure it breaks or deforms at the lowest ultimate crash (impact) load as long as the minimum working load criterion of § 29.952(d)(1) is also satisfied. It should be noted that the ratio of the ultimate failure load of the weakest component in the frangible or deformable component's load path and the normal service load (i.e., the peak load or approved clipped peak load experienced during a typical flight) of that component should be as

high as possible and still meet the other load criteria of this section. Typically this ratio should not be less than 5. The following certification tests (reference MIL-STD-1290) or equivalent should be conducted on each frangible or deformable attachment design.

(i) Static Tests. Each frangible or deformable device should be tested in the three most likely anticipated modes of failure as defined in paragraph d(13). Test loads should be applied at a constant rate not exceeding 20 inches per minute until failure occurs.

(ii) Dynamic Tests. Each frangible or deformable attachment should be tested under dynamic loading conditions. The attachment should be tested in the three most likely failure modes as determined in paragraph d(13). The test load should be applied in less than 0.005 second, and the velocity change experienced by the loading jig should be 36 ±3 feet per second. It should be noted that the dynamic load pulse is a ramp function starting at either 0 or some small test fixture preload and reaching the previously determined failure load in 0.005 seconds. The velocity change of the test jig is also a ramp function starting at 0 and reaching a final velocity of 36±3 ft./sec. in 0.005 seconds. These ramps functions simulate the dynamic conditions of a survivable impact under which the frangible/deformable attachment must perform its intended function.

(15) Section 29.952(d)(3) requires a frangible or locally deformable attachment to meet the fatigue requirements of § 29.571 to eliminate premature fatigue failure. The simplified method of AC 20-95 may be used. Because of service history, all fatigue sources (especially high-cycle vibratory sources) should be reviewed. Fracture critical materials should be avoided and ductile, damage tolerant materials utilized.

(16) Section 29.952(e) requires that, as far as practicable, fuel and fuel containment devices be adequately separated from occupiable areas and potential ignition sources. Several generic categories of ignition sources and potential PCF-producing contact scenarios exist. The intent of the section is to define all possible leak and ignition sources that could be activated in a survivable impact and to provide design features to eliminate or minimize them such that the occurrence of PCF is minimized and escape time is maximized. Adequate separation should be accomplished by a thorough design review, potential PCF hazard analysis, and detailed design trade studies. The resultant findings should be documented and approved. The following PCF hazards and any other such hazards should be documented, minimized by design to the maximum practicable extent, and their resolution documented and FAA/AUTHORITY approved. Conditions to be reviewed should include, but are not limited to, the following:

(i) High temperature ignition sources.

(A) Tank fillers or overboard fuel drains should not be located adjacent to engine intakes or exhausts so that fuel vapors could be ingested and ignited.

(B) Fuel lines should not be located in any occupiable area unless they are shrouded or otherwise designed to prevent spillage and subsequent ignition during and immediately following a survivable impact.

(C) Fuel tanks should not be located in or immediately adjacent to engine compartments, engine induction or exhaust areas, heaters, bleed air ducts, hot air-conditioning ducts, or any other hot surface.

(D) Fuel lines should be kept to a minimum in the engine compartment. Fluid lines should not be located immediately adjacent to engine exhaust areas, heaters, bleed air ducts, hot air-conditioning ducts, or any other hot surface.

(E) Fuel lines should not be located where they can readily spill, spray, or mist onto hot surfaces or into engine induction or exhaust areas. These locations should be determined for each aircraft design by considering probable structural deformation hazards in relation to the fuel system.

(ii) Electrical ignition sources.

(A) Fuel tanks and lines should not be located in electrical compartments.

(B) Electrical components and wiring should be separated from fuel lines and vent openings kept to a minimum in fuel areas.

(C) Electrical wiring should be hermetically sealed, and equipment should be explosion proofed in areas where they are immersed in or otherwise directly subjected to fuel and vapors and should meet § 29.1309 or should be otherwise protected such that ignition is extremely improbable.

(D) Electrical sensor lines that penetrate fuel tank walls should be protected from abrasion or guillotine cutting during a survivable impact by use of potting, rubber plugs or grommets, or other equivalent means and should be designed with sufficient local slack, or equivalent means, to prevent both the wires and their protective mountings from being cut by or torn from fuel tank walls by local deformation.

(E) Electrical wires should be designed with sufficient slack or equivalent means to accommodate structural deformation without creating an ignition source.

(F) Electrical wires that could be subjected to severe local abrasion, cutting, or other damage during a survivable impact should be protected locally by nonconductive shields or shrouds.

(G) Electrical wires that are not sufficiently separated from heat or ignition sources to avoid potential contact during a survivable impact should be locally shrouded with a nonconductive fireproof shroud.

(iii) Friction spark, chemical, and electrostatic ignition sources. Fuel lines and tanks should be designed and located to eliminate fuel or fuel vapor ignition from potential mechanical friction spark ignition sources, chemical ignition sources, and electrostatic ignition sources having a high probability of being activated or created during a survivable impact.

(iv) Separation of fuel tanks and occupiable areas. Fuel tanks should be located as far as practicable from all occupiable areas. This minimizes potential PCF sources in occupiable areas and the potential for occupant saturation with fuel on impact. The design should be reviewed to minimize these potential hazards. Fuel tanks should also be removed, as far as practicable, from other potentially hazardous areas such as engine compartments, electrical compartments, under heavy masses (e.g., transmissions, engines, etc.), over landing gear, and other probable areas of significant impact damage, including rollover and skidding damage.

(v) Fuel Line Shielding. Areas of the fuel line system where the probability of spilled fuel reaching potential ignition sources or occupiable areas is greater than extremely improbable should be shielded with drainable fireproof shrouds. Shrouds should be drainable to allow periodic inspections for internal fuel leaks. The design should be reviewed to ensure these criteria are met.

(vi) Flow Diverters and Drain Holes.

(A) Drainage holes should be located in all fuel tank compartments to prevent the accumulation of spilled fuel within the aircraft. Holes should be large enough to prevent clogging by typical debris and to prevent fluid accumulation from surface tension force blockage.

(B) Drip fences and drainage troughs should be used to prevent gravity-induced flow of spilled fuels from reaching any ignition sources such as hot engine areas, electrical compartments, or other potential hot spots. Drip fences and troughs are also necessary to prevent PCF by routing spilled fuel around ignition sources to drainage holes to minimize fuel accumulation inside the fuselage. Recurring inspection requirements to ensure holes and troughs remain airworthy should be identified. These criteria should be met, as far as practicable, for all postcrash attitudes. This is readily accomplished for the standard landing attitude, but is more difficult for other abnormal attitudes. However, the design should be thoroughly reviewed to insure maximum compliance without adversely impacting other safety and design criteria such as aerodynamic smoothness.

(vii) Fuel Drain System. The fuel drain system and its attachments to the airframe should be designed and constructed, as far as practicable, to be crash resistant. The following and other appropriate means should be considered for a crash resistant design. Tank drains should be recessed or otherwise protected so that they are minimally damaged by impact. Attachment of fuel drains to the airframe should be made with either frangible fasteners or equivalent means to prevent impact induced tearout and leakage. The number of drains should be minimized by design techniques such as those that avoid low points in the lines. Drain lines should be made of ductile materials or otherwise designed to provide impact tolerance. Drain line connections, fittings, and other components should be designed to meet the fatigue requirements of § 29.571 and § 29.952(d)(3). This ensures that unintended partial or full fatigue failures do not occur in normal operations that, if undetected, could compromise the CRFS's intended level-of-safety for the mitigation of post crash fire in a survivable impact. Drain valves should be designed to have positive locking provisions in the closed position in accordance with § 29.999(b)(2).

(17) Section 29.952(f) specifies that fuel tanks, fuel lines, electrical wires, and electrical devices must be designed and constructed, as far as practicable, to be crash resistant. Typical mechanical design criteria necessary to minimize fuel spillage sources, ignition sources, and their mutual contact in a survivable impact (i.e., provide crash resistance) are stated by the following subparagraphs. These mechanical design criteria should be incorporated in each design to the maximum practicable extent. Compliance is accomplished and assessed by a thorough design review and potential PCF hazard analysis with findings and solutions that are documented and approved. Any additional PCF hazards that are identified should be documented, included, addressed equally, and eliminated to the maximum practicable extent. Engineering evaluation, analysis, and tests are all required to determine the maximum level of practicability.

(i) They should not initiate or contribute to a post crash fire in an otherwise survivable impact. A hazard analysis should show which components are critical in this regard and should be assessed in detail for hazard elimination purposes.

(ii) Fuel and electrical lines and components should be located away from each other, away from probable crash impact areas, and away from areas where structural deformation or large objects (such as engines or transmissions) may, by crushing or penetration, cause fuel spillage or create an electrical ignition source, or both.

(iii) Fuel and electrical lines and components should be located separately and away from areas where impact and severing by rotor blades during a survivable impact are probable.

(iv) Fuel and electrical lines and components should be in no danger of being punctured or severed during a survivable impact by locally stiff vertical understructure such as a collapsed landing gear strut.

(v) Fuel and electrical lines and components should be routed separately in areas of maximum protection, such as along heavier structural members, and away from areas where significant damage is probable.

(vi) Fuel and electrical lines and components running through hazardous areas or directly through structure, such as a bulkhead, should be locally separated and protected from over-extension, severe abrasion and guillotine cutting by frangible panels, suitable clearance, rubber grommets, braided armor shielding (which should be nonconductive for electrical lines), or other equivalent means.

(vii) Fuel lines routed directly to instruments, transducers, or other equivalent devices should be crash resistant, in accordance with § 29.1337(a)(2), to minimize leakage in case of line rupture induced during a survivable impact.

(viii) Electrical wires routed directly into electrical boxes or instruments should be designed with sufficient local slack and locally routed in the least probable damage direction and zone, or otherwise protected to minimize the probability of damage-induced arcing.

(ix) Fuel lines routed directly into fuel tanks or other fuel system components should be locally routed in the least probable damage direction and zone, or otherwise protected, to minimize the probability of damage-induced fuel leaks.

(x) Fuel pumps mounted inside fuel tanks should be rigidly attached to the fuel tank only. If the pump is airframe mounted and has structural significance, it should have a frangible or deformable attachment (reference paragraph d(12)). Electrical boost pumps, if used, should be installed with a minimum of 6 inches of slack wire at the pump connection. The pump wires should be shrouded to prevent cutting in a survivable impact. Nonsparking, breakaway wire disconnects or other equivalent means may be used in lieu of the 6 inches of slack wire.

(xi) Fuel filters and strainers, to the maximum practicable extent, should not be located in or adjacent to the engine intake or exhausts and should retain the smallest practicable quantity of fuel.

(xii) The number of fuel valves should be kept to a minimum. If electrically operated valves are used, they should be installed with a minimum of 6 inches of slack in the electrical lines, unless protected by equivalent means (reference 17(i)). The valves should be installed with the maximum amount of protection and separation of the electrical wires from the remainder of the valve assembly.

(xiii) Fuel quantity indicators mounted in or on fuel tanks should be selected, designed, and installed to provide the minimum puncture or tear hazard to the fuel tank in a survivable impact.

(xiv) Fuel tank and bladder enclosures should have smooth, regular shapes that avoid sharp edges and corners. Minimum concave and convex radius design criteria should be developed and adhered to. Magnesium should not be used in fuel cells, and any cadmium-plated parts should not be exposed to fuel.

(xv) Any shielding of electrical wires from abrasion, cutting, or overextension must be nonconductive.

(xvi) All fuel line installations not containing breakaway couplings should be reviewed to insure that they will not be overtensioned in a survivable impact, that they are properly grouped and properly exit fuel tanks, firewalls, and bulkheads in the area of least probable damage, and that their number and lengths are safely minimized.

(xvii) Crash resistance guidance for other basic components is contained in related paragraphs such as AC 29.963 (§ 29.963, bladders and liners), AC 29.973 (§ 29.973, fuel tank filler connections) and AC 29.975 (§ 29.975, fuel tank vents).

(18) Section 29.952(g) requires rigid or semirigid fuel tank or bladder walls of any material construction to be both impact and tear resistant. This minimizes a PCF from impact-induced rupture and tear.

(i) A rigid tank or bladder can resist fluid pressure loads as a flat plate in bending. A semirigid tank can resist fluid pressure loads partially as a flat plate in bending and partially as a membrane in tension. Flexible liners are exempt from the requirements of § 29.952(g) since an unsupported flexible liner can resist only pure tension loads acting as a membrane (i.e., it has negligible bending strength). The rigid shell structure required by § 29.967(a)(3) that surrounds the flexible liner (membrane) carries the crash-induced impact and tear loads; whereas, the flexible liner is only significantly loaded in tension if the shell structure is penetrated by a sharp object on impact.

(ii) For metallic tanks, rigid or semirigid composite tanks (resin matrix), semirigid bladder designs (rubber matrix), metal-composite hybrid designs, and all other tank designs, impact and tear resistance should be shown by analysis and tests.

(iii) Designs using resin matrix composites should be subjected to the composite structure substantiation guidance of AC 20-107A, Composite Aircraft Structure, dated April 25, 1984, and paragraph AC 29 MG 8. Designs using rubber matrix composites are subject to the standard substantiation requirements for these devices, such as TSO-C80.

(iv) One set of crash resistance tests that constitutes an acceptable method of substantiation to the requirements of § 29.952(g) for all tank designs regardless of the materials used are those specified in paragraphs 4.6.5.1 (Constant Rate Tear); 4.6.5.2 (Impact Penetration); 4.6.5.3 (Impact Tear); 4.6.5.4 (Panel Strength Calibration); and 4.6.5.5 (Fitting Strength) of MIL-T-27422B, “Military Specification; Tank, Fuel, Crash-Resistant Aircraft.” These test requirements, or equivalent means, should be applied for and discussed early in certification. If the MIL-T-27422B tests are selected, severity differences between military combat requirements and the civil environment should be accounted for by reducing the MIL-T-27422B requirements, as follows:

(A) Constant Rate Tear. The minimum energy for complete separation should be 200 foot-pounds (reference 4.6.5.1).

(B) Impact Penetration. The drop height of a 5-pound chisel should be reduced to 8.0 feet (reference 4.6.5.2).

(C) Impact Tear. The drop height of a 5-pound chisel should be reduced to 8.0 feet and the average tear criteria should not exceed 1.0 inch (reference 4.6.5.3).

(19) Section 29.952(g) also requires that all fuel tank designs (regardless of the materials utilized and whether or not a flexible liner of any type is used) for each tank or the most critical tank be analyzed and tested to the criteria of § 29.952 d(18)(iv), or equivalent.

(20) Any type of flexible liner or bladder used in any type of fuel tank construction (integral, hard shell, etc.) must meet the strength and puncture resistance requirements of § 29.963(b). Section 29.963(b) contains the new puncture resistance requirement for flexible liners and other liner material certification requirements. Unlined, bladderless fuel tanks are also required to meet this requirement. Most unlined, rigid fuel cell designs should readily exceed the 370-pound minimum puncture force requirement because of overriding design requirements and material characteristics, such as stiffness and ductility.

NOTE: TSO-C80, "Flexible Fuel and Oil Cell Material," is referenced in the advisory material for § 29.963(b) and contains the detailed qualification requirements for these materials. The current puncture resistance test of TSO-C80, paragraph 16.0, states that the force required to puncture the bladder material must be greater than or equal to 15 pounds (e.g., screwdriver test). Section 29.963(b) has increased the TSO paragraph 16.0 puncture force value to be greater than or equal to 370 pounds. This is for fuel cell bladder or liner material only. Oil cell material puncture force requirements are not changed.

e. Typical Examples of Loading Modes and Test Setups for CRFS Components. The following figures, which are referred to periodically in the advisory circular, show typical examples of test setups for CRFS components such as breakaway fuel fittings, hoses, hose end fittings, and hose assemblies.

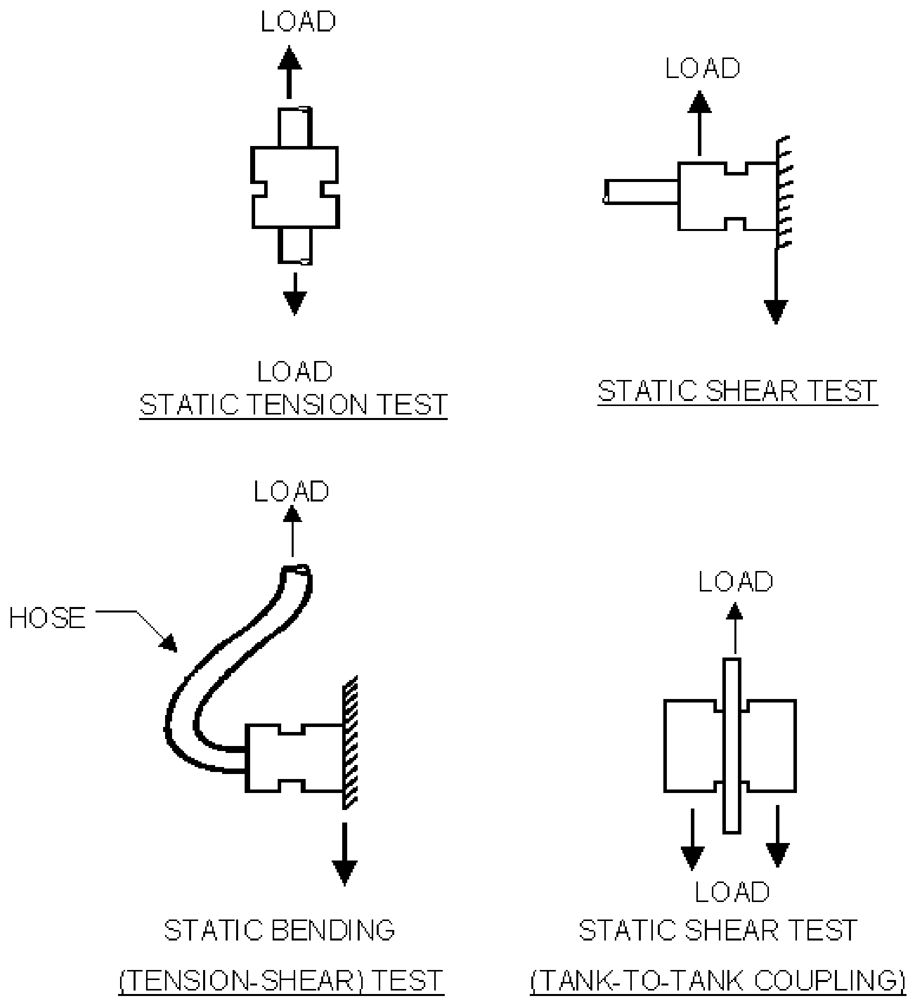


FIGURE AC 29.952-1 STATIC TENSION AND SHEAR LOADING MODES

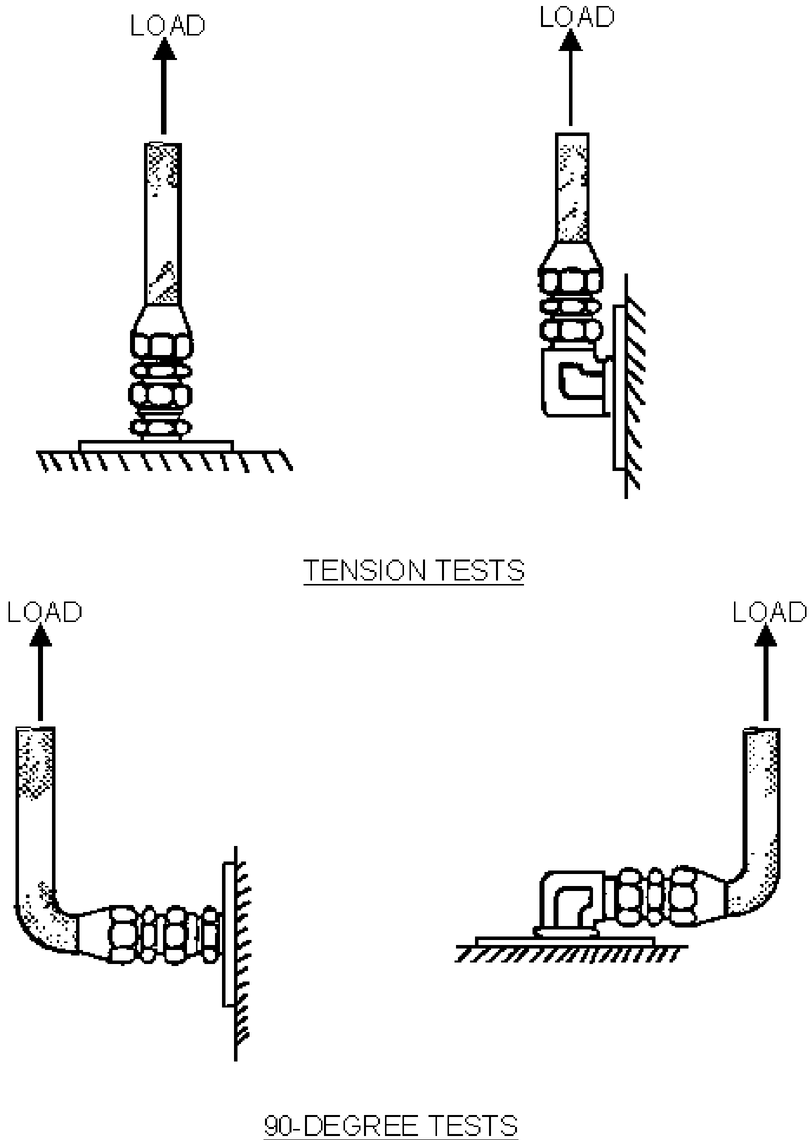
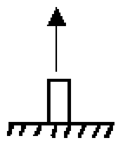
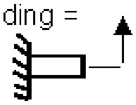
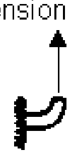
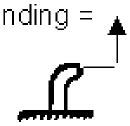
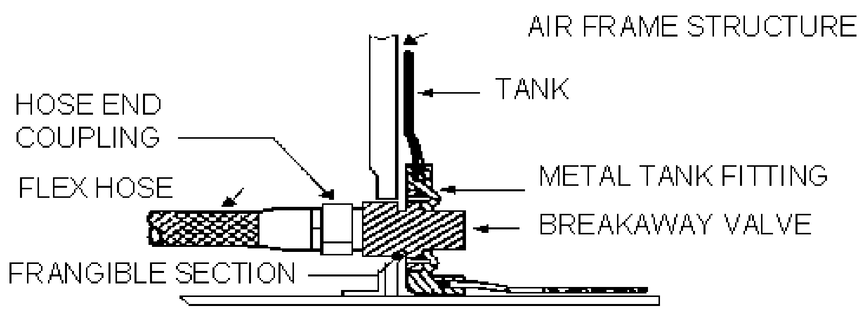


FIGURE AC 29.952-2 HOSE ASSEMBLY TESTS

Hose End Fitting Type	Fitting Size	Tension Load (lb)		Bending Load (lb)	
		Minimum Average Load*	Minimum Individual Load	Minimum Average Load*	Minimum Individual Load
STRAIGHT	-4	600	475	425	400
Tension =	-6	700	575	425	400
	-8	900	650	650	600
	-10	1450	1175	675	625
Bending =	-12	1775	1475	950	850
	-16	2125	1825	1425	1300
	-20	2375	2075	1550	1425
90° ELBOW	-4	600	475	425	400
Tension =	-6	700	575	425	400
	-8	900	650	450	400
	-10	1450	1175	475	425
Bending =	-12	1775	1475	500	450
	-16	2125	1825	775	700
	-20	2375	2075	1100	1000

*Average of at least 3 tests.

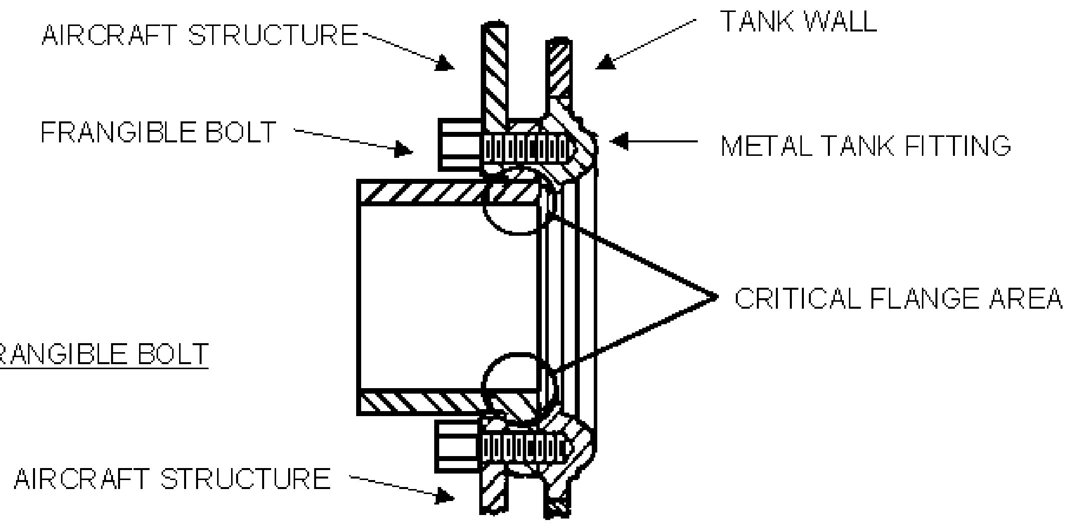
FIGURE AC 29.952-3 MINIMUM AVERAGE AND INDIVIDUAL LOADS FOR HOSE AND HOSE-END FITTING COMBINATIONS



ITEM	LOWEST FAILURE LOAD (LB)*	FAILURE MODE
Flex Hose	**3000**	**Tension Breakage**
Flex Hose	**1500**	**Pull Out of End Fitting**
Tank Fitting	**7500**	**Pull Out of Tank**
Hose End Coupling	**1650**	**Break (Bending)**
Breakaway Valve	**2500**	**Pull Out of Tank Fitting**
Breakaway Valve	**Not More Than** $\frac{1500}{2} = 750$ **Not Less Than** $\frac{1500}{4} = 375$	**Break at frangible Section**

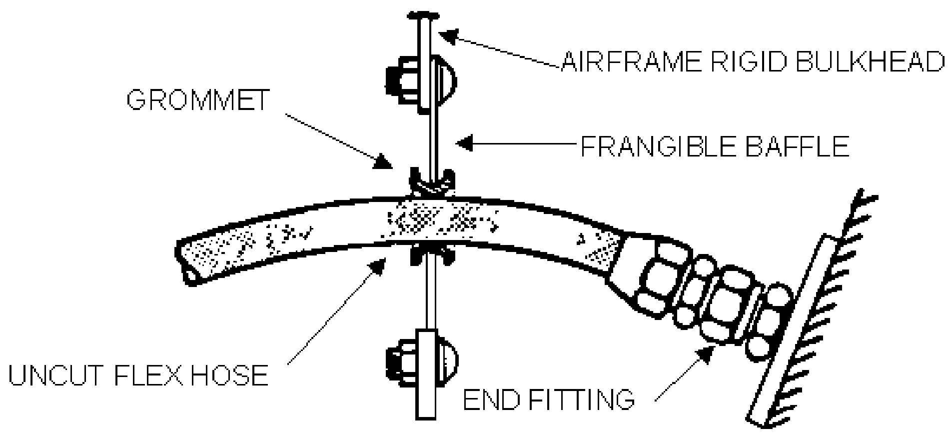
*Loads may or may not be representative; values are for explanatory purposes only.

FIGURE AC 29.952-4 TYPICAL METHOD OF BREAKAWAY FUEL FITTING LOAD CALCULATIONS (TANK INSTALLATION USED AS EXAMPLE ONLY; BASIC TECHNIQUE APPLICABLE TO OTHER CONFIGURATIONS)



ITEM	LOWEST FAILURE LOAD (LB)*		FAILURE MODE
AIRCRAFT STRUCTURE	4000		SHEAR
TANK FITTING	3000		PULLOUT OF TANK
FLANGE	5000		SHEAR
FRANGIBLE BOLT	NOT MORE THAN $\frac{3000}{2} = 1500$	NOT LESS THAN $\frac{3000}{4} = 750$	BREAK (TENSION-SHEAR)

FIGURE AC 29.952-5 TYPICAL METHODS OF FRANGIBLE OR DEFORMABLE ATTACHMENT LOAD CALCULATIONS: EXAMPLE 1, FRANGIBLE BOLTS.



ITEM	LOWEST FAILURE LOAD (LB)*		FAILURE MODE
RIGID BULKHEAD	4000		BEARING
FLEX HOSE	3000		TENSION BREAKAGE
FLEX HOSE	1500		PULLOUT OF END FITTING
END FITTING	1750		BENDING
FRANGIBLE BAFFLE	NOT MORE THAN $\frac{1500}{2} = 750$	NOT LESS THAN $\frac{1500}{4} = 375$	BEARING

*VALUES ARE SHOWN FOR EXPLANATORY PURPOSES ONLY

FIGURE AC 29.952-6 TYPICAL METHODS OF FRANGIBLE OR DEFORMABLE ATTACHMENT LOAD CALCULATIONS: EXAMPLE 2, FRANGIBLE BAFFLE.

AC 29.953. <u>§ 29.953 FUEL SYSTEM INDEPENDENCE</u>.

a. <u>Explanation</u>.

(1) Section 29.953(a)(1) stipulates that fuel systems for Category A rotorcraft must meet the requirements of § 29.903(b) engine isolation.

(2) Section 29.953(a)(2) specifies independent fuel feed systems for each engine for Category A rotorcraft unless other provisions are made to meet the § 29.903(b) engine isolation requirement.

(3) Section 29.953(b) specifies independent fuel feed systems for each engine for Category B rotorcraft, except that separate fuel tanks are not required.

b. <u>Procedures</u>.

(1) The purpose of § 29.953(a) is to ensure an independent fuel supply system for each engine. Multiengine Category B rotorcraft do not require separate fuel tanks, as are intended for Category A.

(2) The assessment to ensure compliance with § 29.903(b), engine isolation, should include consideration of component failure, malfunction, and damage. For multiengine Category B rotorcraft, leakage of the fuel cell could be excluded from consideration since § 29.953(b) explicitly states that separate fuel tanks are not required for this category rotorcraft.

NOTE: Of interest is that § 29.903(c), engine isolation for normal category airplanes, also excludes the fuel tank from consideration if only one tank is used.

(3) Consideration of fuel tank leakage under § 29.903(b) has dictated separate fuel tanks for Category A rotorcraft, but the regulation leaves the door open for unique designs by the expression, “Unless other provisions are made…,” in § 29.953(a)(2). Separate tanks are intended for Category A as evidenced by the identical fuel system independence requirements for multiengine Category B rotorcraft, except that separate tanks are specifically not required.

(4) A common supply tank, with individual “collector” tanks for each engine for Category A rotorcraft, has been allowed under § 29.953 provided that the capacity of the collector tanks will allow 20 minutes of maximum allowable en route OEI power.

(5) The fuel system independence regulations are not intended to preclude single-point fueling designs.

(i) For multiengine Category B rotorcraft, the assessment of an independent fuel supply system for each engine would begin at the fuel supply pickup point within the tank and continue to the engine fuel inlet at the engine.

(ii) For Category A rotorcraft, the assessment would begin with the tanks and continue to the engine fuel inlet.

(6) If supply line crossfeed capability is included as a feature, care must be exercised to ensure that the opening of the crossfeed does not jeopardize the continued safe operation of more than one engine. For example, if the crossfeed valve is automatically operated by a low pressure signal in the supply line for one engine, the possibility that fuel line leakage could cause opening of the crossfeed and jeopardize the continued safe operation of both engines should be considered. Similarly, opening the crossfeed valve with a suction lift system following engine or system malfunction should not allow air into the fuel supply line of the remaining engine.

(7) The independent fuel supply system requirement for each engine is for normal fuel system operations. Care should be exercised to ensure that flight manual procedures do not authorize normal usage of fuel system configurations which may violate the engine isolation principle. For example, routine fuel balance procedures should not allow usage of a common supply line if a failure can jeopardize the continued safe operation of more than one engine.

(8) Fuel system designs which allow the continued safe operation of all engines under expected fuel system component failure conditions (for example, a failed boost pump) by using common fuel flow paths under failure conditions are not prohibited.

(9) For APU's which perform a required in-flight function, a separate, independent fuel system complying with the corresponding engine fuel system rules should be provided. Other APU's (which do not perform a required in-flight function) may be supplied with fuel from a tee connection to a main engine fuel supply. The fuel shutoff valve for the APU should be located as close as possible to the APU system's connection to the main engine fuel system and a checkvalve should be included in the APU fuel system to prevent reverse-flow if negative pressure exists momentarily in the main engine fuel system. Maximum fuel demand of the APU will not jeopardize compliance with § 29.955.

AC 29.954. § 29.954 (Amendment 29-26) FUEL SYSTEM LIGHTNING PROTECTION.

a. Background. During the initial development and promulgation of the standards concerning the airworthiness of rotorcraft, it was not deemed necessary to specify design features that would protect the rotorcraft from the meteorological phenomenon of lightning. This was due, in part, to the fact that rotorcraft were primarily operated in a VFR and non-icing environment. Also, a prudent pilot avoided thunderstorms where the possibility of encountering severe weather and a lightning strike was much greater. The construction, design, and operating environment of civil rotorcraft have changed markedly within the past two decades. Many rotorcraft are now authorized to fly IFR in all types of weather environment. One transport design has been approved for flight into known icing conditions. Additionally, many rotorcraft now use the same advanced technologies in structures and systems as do airplanes. Because of these facts the possibility of a lightning strike encounter to the rotorcraft has been greatly increased. If the fuel system of the rotorcraft has not been properly designed and constructed, a fuel vapor ignition may occur. This occurrence generally results in a catastrophe to the rotorcraft. To prevent such a catastrophe and provide a level of safety equivalent to transport category airplanes, a specific rule for the lightning protection of transport category rotorcraft fuel systems was adopted in Amendment 29–26.

b. Explanation.

(1) This regulation requires that the rotorcraft's fuel system be designed and constructed so that an ignition of fuel vapor will not occur when the rotorcraft is involved in a lightning strike. For the purposes of this regulation the fuel system is comprised of the fuel tank with all its associated plumbing and any other areas of the rotorcraft likely to have fuel vapor present (such as sumps and drains for the tank itself). Externally mounted fuel tanks are also considered to be part of the "fuel system."

(2) Other associated installations such as electrical wiring in the fuel tanks which could provide a source of ignition due to an indirect or induced effect should also be considered.

c. Procedures.

(1) The current revision of Advisory Circular 20-53 provides guidance on an acceptable method and procedure to be utilized to demonstrate that the design and construction of the fuel system is compliant with § 29.954.

(2) FAA Report No. DOT/FAA/CT-89/22 contains additional information regarding the lightning environment. Also contained in this report are design and test techniques which provide for a design that will be adequately protected from fuel vapor ignition when the rotorcraft encounters the lightning environment. This report is available to the public by order from the National Technical Information Service, Springfield, VA 22161.

AC 29.955. § 29.955 (Amendment 29-2) FUEL FLOW.

a. Explanation.

(1) Section 29.955 is intended to ensure adequate fuel flow to the engine(s) at maximum power under the intended aircraft operating conditions and maneuvers. In ensuring adequate fuel flow, both hot and cold fuel would normally be evaluated for the suction lift system, whereas cold fuel is usually more critical for the boosted pressure system.

(2) In showing adequate fuel flow, the rule provides that--

(i) The fuel be supplied within the appropriate engine fuel pressure range;

(ii) The test be conducted with minimum fuel onboard, consistent with test safety;

(iii) For pump systems, fuel flow requirements be satisfied with the critical airframe furnished pump inoperative; and

(iv) The fuel flowmeter, if installed, must be blocked such that fuel must flow through the meter or its bypass.

(3) Section 29.955(b) specifies that if an engine can be supplied with fuel from more than one tank, the fuel system must feed promptly when fuel becomes low in one tank and another tank is selected.

b. Procedures.

(1) Testing (including bench tests) has been the accepted method to show compliance with § 29.955(a). Analytical techniques may be used to adjust the system test results to various fuel conditions and flows or to account for minor modifications to a system. A purely analytical approach is not generally acceptable.

(i) Methods to adjust the test data for different fuel properties and flows should be verified by limited testing.

(ii) If a suction lift system is used and hot fuel verification is involved (reference § 29.961) testing is appropriate.

(2) Demonstrating that the system is capable of providing "…100 percent of the fuel flow required under the intended operating conditions…" will depend on the particular system design, whether boosted or suction lift, Category A or Category B, and whether single or multiengine. Some of the factors to be evaluated are as follows:

(i) Acceleration fuel flow requirements may exceed those for steady-state operation. For example, if on a cold day, engine torque is the limiting parameter, the steady-state fuel flow demand corresponding to that torque may be exceeded during engine acceleration to that power.

(ii) For single-engine rotorcraft and for multiengine rotorcraft with all engines operating, some margin should be included to account for possible inadvertent overtorque.

NOTE: Notice of Proposed Rulemaking (NPRM) No. 84-19 proposes to include this consideration as a firm requirement (reference 49 FR 46670; dated November 27, 1984).

(iii) For multiengine rotorcraft, adequate fuel flow under OEI conditions should be ensured.

(A) For Category A systems, evaluation of § 29.903(b) should ensure that following the failure of one engine, lack of fuel flow will not jeopardize the safe operation of the remaining engine(s). Since governor-controlled engines will automatically accelerate to some limit if power demand is high, and since immediate crew action is not presumed under § 29.903(b), compliance with § 29.955 would include adequate fuel flow to the cold day maximum OEI torque to be expected (reference § 29.927(b)(2)).

(B) A proposed revision to § 29.955 (reference NPRM No. 84-19) would require that fuel flow for multiengine Category B rotorcraft be adequate for the § 29.927(b)(2) OEI overtorque condition.

(C) Following an engine failure, the remaining engine(s) may accelerate to the gas producer speed topping limit fuel flow, rather than to the fuel flow for the steady-state OEI power value. This consideration would be most important for suction lift systems which may be critical with hot fuel at altitude.

(3) The critical fuel system configuration should be evaluated.

(i) For pump fed (boosted) systems, fuel flow requirements should be satisfied with the critical airframe furnished pump inoperative.

(ii) If on multiengine rotorcraft it is acceptable to operate following an engine failure in more than one fuel system configuration (for example, if crossfeed is an acceptable mode), then the supplying of multiple engines through common components may be more critical than the OEI condition.

(4) Adverse transient and steady-state maneuver loads should be considered since the g-loading experienced may tend to decrease the engine fuel inlet pressure below allowable limits.

(5) The fuel should be delivered to the engine inlet within the limits specified in the engine type certificate. The method of specifying these fuel inlet pressure requirements varies with the engine model. Some of these include:

(i) Specification of a gage pressure as a function of altitude for suction system operation. The particular fuel and fuel temperature for demonstrating the criteria may be specified in the engine documents. Other approved fuels, fuel temperatures, and boost-pump-on operation are considered satisfactory if the demonstration with the specified fuel is successful.

(ii) Specification of a maximum allowable vapor-to-liquid ratio for hot fuel, and minimum absolute pressure as a function of altitude for cold fuels.

(iii) Specification of a fuel inlet pressure relative to the true vapor pressure of the fuel, in combination with a maximum allowable vapor-to-liquid ratio.

(iv) Specification of separate pressure limits for boost-on and suction lift operation.

(v) Specification of special limits for emergency use or emergency fuels.

(6) For those systems which specify a minimum V/L ratio, the methods provided in Aerospace Recommended Practice (ARP) 492 published by the Society of Automotive Engineers are acceptable in evaluating test results.

(7) Since the lower quantity of fuel in the tank will reduce the hydrostatic head and thus the fuel inlet pressure, § 29.955(a)(2) specifies that the quantity of fuel in the tank should be minimum.

(8) Section 29.955(a)(3) specifies that each main and emergency pump be evaluated. If it can be determined which pump and flow path is critical, only that configuration would be tested. Similarly, for suction fuel systems, the critical flow paths and flow requirements should be evaluated. If pumps are required to supply the necessary fuel, § 29.1305 would require a fuel pressure indicator and § 29.1549 would require a red radial at the minimum safe operating fuel pressure for any fuel or fuel usage condition. This pressure limit should be used to determine compliance with § 29.955(a)(1) for all operations.

(9) Section 29.955(a)(4) specifies that the fuel flowmeter, if installed, be “blocked” in showing compliance with the fuel flow requirements. Consideration of flowmeter component failure or malfunction would most often be more appropriate than blockage.

(i) If the flowmeter is completely blocked in assessing compliance, then a bypass would be dictated, and the provision for “flow through the meter” following blockage would not be a viable alternative. It is not the intent of the rule to arbitrarily preclude flowmeter installations without a bypass system.

(ii) Section 29.1337(c) clarifies that if the <u>malfunction</u> of a metering component severely restricts fuel flow, a bypass would be required. An example of a malfunction to be considered would be a locked rotor on a rotating element design.

(iii) NPRM No. 84-19 proposes to clarify the intent of § 29.955 by requiring that proper fuel flow be ensured with fuel flow transmitter component failure, rather than with transmitter blockage as specified in the existing rule.

(10) Section 29.955(b) requires the fuel system to feed promptly when fuel becomes low in one tank and another tank is selected. This requirement is important because momentary fuel flow interruption must be expected to result in complete power failure and, for single engine rotorcraft, an emergency landing.

AC 29.955A. <u>§ 29.955 (Amendment 29-26) FUEL FLOW</u>.

a. <u>Explanation</u>. Amendment 29-26 adds new requirements for test conditions to ensure that adequate fuel flow is available to the engine in critical combinations of adverse conditions that may be expected during operation of the rotorcraft. The amendment also requires a correlation between fuel filter blockage and the fuel filter warning device required by § 29.1305(a)(17). Design and performance standards for auxiliary fuel tank and transfer tank fuel systems are provided. These changes were made to ensure that all parameters associated with fuel supply to the engine are adequately addressed.

b. <u>Procedures</u>.

(1) Section 29.955 is intended to ensure adequate fuel flow to the engine(s) during all operating conditions of the rotorcraft. This includes the fuel flows necessary to operate the engine(s) under the test conditions required by § 29.927. Testing (including bench or rig tests) has been the accepted method of showing compliance with this section although analytical techniques may be used to adjust system test results to various fuel flow conditions or to account for minor modifications to a system. Analytical methods that are used to adjust the test results should be verified with limited testing. It should be shown during compliance testing that the fuel pressure, at the engine to airframe interface, will be within the limits specified by the engine manufacturer. The fuel pressure at this point should be maintained within limits specified by the engine manufacturer during all critical maneuvers and accelerations. All of the following conditions should be met during compliance testing unless it can be shown that combinations of the conditions are not possible.

(i) The fuel quantity in the tank(s) in use during the test may not exceed the unusable fuel quantity established under § 29.959, plus the minimum quantity required to conduct the test.

(ii) During the compliance test, the rotorcraft should be maneuvered to create the most critical fuel pressure head between the fuel tank outlet and the engine to airframe interface (engine fuel inlet).

(iii) For boost pump fed systems, it should be determined which pump (primary or secondary) would create the most critical restriction if it failed. The critical pump should then be installed to create the critical restriction, either by actual or simulated failure.

(iv) Various combinations of engine power demand, electrical power available, and motive flow requirements for ejector pumps, will have an effect upon the fuel flow and pressure available at the engine to airframe interface. Adequate fuel pressure should be available to the engine with the most critical combination of these parameters.

(v) Critical values of fuel properties that may adversely affect fuel flow and/or fuel pressure should be applied. This includes alternate types of fuel if certification with alternate fuels is requested. At the minimum, the fuel that will create the highest vapor to liquid ratio should be used during hot fuel tests (§ 29.961). The most viscous fuel should be used during cold fuel tests.

(vi) The fuel filter, required by § 29.997, should be partially blocked to simulate the maximum contamination allowable. The blockage should be sufficient to activate the impending bypass indicator that is required by § 29.1305(a)(17).

(2) Unique Conditions. The phrase, "…Provide the engine with at least 100 percent of the fuel required under all operating and maneuvering conditions…" (§ 29.955(a)), includes unique flight conditions within the operational envelope of the rotorcraft. Critical conditions of fuel flow to the engine(s) may exist under the following conditions (and others identified by the applicant); therefore, they should be evaluated and tested if applicable:

(i) In a single engine rotorcraft, a rapid acceleration to maximum power (torque) that will be requested for certification may be a critical condition. In this case the fuel flow required during the transient may exceed the fuel flow required for steady state at the maximum power condition.

(ii) In multiengine rotorcraft, a rapid acceleration to the maximum OEI power rating that will be requested may be a critical condition. The fuel flow during the transient may be higher than that required at the steady state OEI condition.

(3) If auxiliary fuel pumps (boost pumps) are used to supply fuel to the engines, and ejector pumps are used for cross-feed or other inter-tank fuel distribution systems, a test should be run that will place the maximum fuel demand on the auxiliary pump(s).

(4) In some multiengine rotorcraft, a single pump may be required to provide fuel flow to all engines in the event of an auxiliary pump failure. If this is the case, a test should be conducted with a simulated (or actual) failed auxiliary pump. If the functional auxiliary pump is designed to provide motive flow for cross-feed systems, the most critical condition of fuel flow demand should be tested.

(5) Transient and steady state maneuver loads (g-loading) may affect the fuel pressure at the engine to airframe interface. This effect should be considered and then tested, if appropriate.

(6) The methods of specifying the engine inlet fuel pressure requirements are sometimes related to fuel temperature and altitude. Therefore, it is necessary to explore the extremes of the envelope to assure compliance rather than attempting to select one critical condition. For instance, the increase in fuel viscosity at cold temperatures may increase system pressure drop and offset a slight drop in required fuel flow. In this case, critical fuel inlet conditions may not be experienced at maximum engine fuel flow.

(7) A conservative demonstration would consider the maximum allowable fuel viscosity in combination with the maximum fuel flow. Otherwise, several test points may be required.

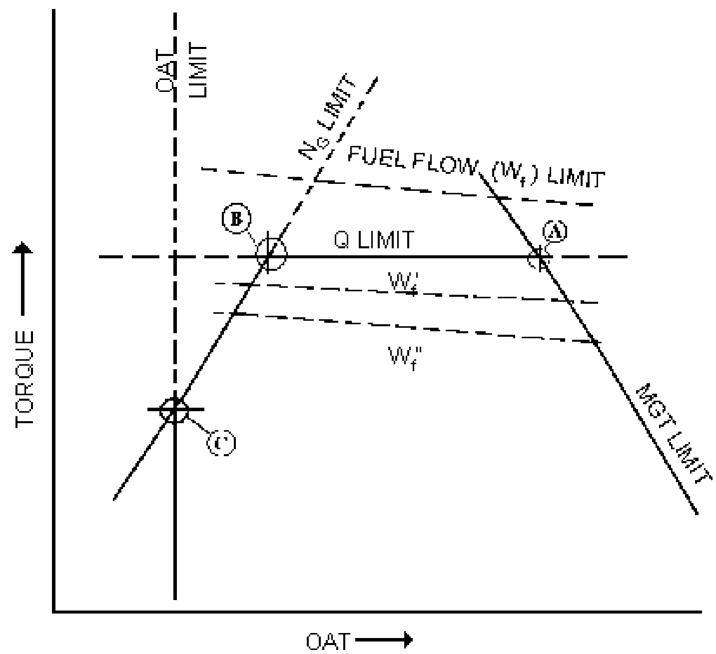


FIGURE AC 29.955A-1 FUEL FLOW

NOTES:

(1) Point A on figure AC 29.955A-1 is the highest fuel flow within aircraft limitations, but the system pressure drop is not expected to be maximum because of the low kinematic fuel viscosity.

(2) Point B is the maximum flow at cold temperatures but as the fuel temperature is further reduced, the fuel viscosity increases very rapidly.

(3) Point C represents the maximum viscosity of the fuel, but the fuel flow is somewhat reduced from point B. The maximum system pressure drops and, therefore, minimum fuel inlet pressure may occur between points B and C depending on the specific relationship of fuel viscosity to required fuel flow.

AC 29.957. <u>§ 29.957 FLOW BETWEEN INTERCONNECTED TANKS</u>.

a. <u>Section 29.957(a)</u>:

(1) <u>Explanation</u>. This paragraph sets forth a design requirement that prohibits approval of a fuel tank interconnect arrangement wherein gravity or acceleration-induced flow between tanks will result in overflow through a tank vent.

(2) <u>Procedures</u>. The design of the vent for the receiving tank should be sufficiently elevated to preclude gravity or flight accelerations from causing overflow through the vent. A flight test may be needed to determine the effectiveness of the arrangement. Check valves in the vent system to prevent overflow should be discouraged because of reliability aspects.

b. <u>Section 29.957(b)</u>:

(1) <u>Explanation</u>. For fuel system arrangements which permit fuel to be pumped from one tank to another, design precautions to prevent structural damage to the receiving tank in the event of overfilling are required as well as a design means to warn the crew before overflow through the vents occurs.

(2) <u>Procedures</u>. The design of the receiving tank should have large vent lines or a recirculation line back to the original tank to prevent overfilling of the receiving tank. Alternatively, a float switch may be used to de-energize the transfer pump, providing that faults in the system do not adversely affect safety. A float switch may be used to warn the crew that overfilling of the receiving tank is impending. If a float switch is used, review the system reliability requirements of § 29.901(c).

AC 29.959. <u>§ 29.959 UNUSABLE FUEL SUPPLY</u>.

a. <u>Explanation</u>. This rule requires the applicant to establish a value for unusable fuel for each tank. This value for unusable fuel may be selected by the applicant to facilitate compliance with § 29.1337(b)(1) provided the amount is equal to or greater than the actual unusable fuel. The actual unusable fuel is the amount of fuel in the tank when, in the critical flight attitude, evidence of system or engine malfunction occurs, or in the case of transfer tanks, when flow to the receiving tank is interrupted.

b. <u>Procedures</u>.

(1) The unusable fuel for each tank can be determined by flight tests which involve flight in the critical attitude or maneuver until indication of a malfunction. For boosted systems, the “first evidence of malfunction” may be a pressure fluctuation to below the fuel pressure minimum redline, engine power fluctuation, or boost pump failure warning indication. For suction lift systems, the indication may be a low fuel pressure warning light. In some instances, particularly for suction-lift systems, special test instrumentation for fuel pressure is required, and, since an accurate measurement

of the remaining fuel in the tank should be obtained, a method to close off flow from that tank would be needed. For transfer tanks, or tanks which are limited to use only during cruise flight, the flight regimes usually can be limited to level flight or hover at the c.g. condition which, by inspection, would create the maximum unusable fuel. For tanks for general use, the flight regimes should also include takeoff and landing using steady pitch attitudes to be expected, as well as hover and level flight conditions. The possible adverse effects of extreme lateral c.g. should be considered.

(2) Normally, these tests are conducted with all equipment (pumps, ejectors, etc.) operating as prescribed by the design. However, values for unusable fuel with pump failures, if significantly different, should also be determined and listed in the flight manual. These values for unusable fuel need not be considered in the empty weight of the aircraft.

c. While the procedures of paragraph b(1) are acceptable, fuel exhaustion during critical flight test conditions must be expected. To minimize this possible flight test hazard, the applicant may in many cases, utilize analysis and/or ground tests involving normally available flight test data on aircraft attitudes, tank configuration studies, and critical flight condition studies to determine unusable fuel. Any questionable results, however, should be resolved by actual flight test or introduction of conservatism into the finding.

AC 29.961. § 29.961 FUEL SYSTEM HOT WEATHER OPERATION.

a. Explanation.

(1) Section 29.961 specifies that a hot fuel test be conducted on suction lift systems, and on other fuel systems conducive to vapor formation, to ensure that the system is free from vapor lock at a fuel temperature of 110° F under critical operating conditions.

(2) Pressure boosted systems would not ordinarily require hot fuel tests unless--

(i) There are high points in the fuel system which would allow accumulation of vapor; or

(ii) The engine fuel inlet pressure is negative relative to tank pressure because of low boost pump pressure or high fuel system pressure losses (but still within fuel pressure limits).

(iii) The airframe boost pump is not actually submerged such that a portion of the system is suction lift.

(3) Boosted system vapor lock difficulties, at relatively low system flows compared to pump capacity, have occurred in at least two instances.

(i) If the fuel pump is a positive displacement type with an internal bypass and the pump capacity significantly exceeds system demand, excessive recirculation within the pump may significantly raise the local fuel temperature resulting in pump cavitation.

(ii) Parallel pump systems, where one supplies the majority of the fuel while the other “deadhead” pump supplies only a negligible amount of fuel, may experience vapor lock and cavitation of the deadhead pump due to excessive recirculation of fuel as described in a(3)(i).

(4) The requirement to use 110° F fuel is a carryover from the recodification of CAR Part 6, although the use of hotter fuel at the same Reid Vapor Pressure would tend more toward vapor formation.

(5) The term “vapor lock” means a change in normal engine operation as a result of the formation of fuel vapor-air mixtures in the fuel feed system.

(6) Section 29.961(b) and (c) inappropriately specify a particular flight condition, weight, and power spectrum which may not be critical. Hence, a demonstration of compliance to the specifics of § 29.961(c) will probably be inadequate for compliance with § 29.961(a)(2). NPRM No. 84–19 proposes to revise § 29.961 to delete these unnecessary, detailed regulations with a simple requirement to show satisfactory operation under critical operating conditions with hot fuel. The guidance which follows should be sufficient to establish compliance with § 29.961, in total, without regard to the misleading specifics of § 29.961(b) and (c).

b. Procedures.

(1) The fuel type to be used should be that with the highest true vapor pressure (TVP) at the 110° F condition.

(2) The fuel should be heated as rapidly as possible since the longer fuel is heated the more vaporization occurs resulting in unconservative test results. Likewise, heating the fuel above the target temperature, then allowing it to cool will “weather” the fuel excessively resulting in a reduction in Reid Vapor Pressure and unconservative testing.

(3) If the test is performed at cool ambients, the fuel lines, tanks, etc., may have to be insulated to ensure that the fuel inlet temperature is approximately the same as would be experienced on a hot day. This should be verified by instrumenting the fuel temperature at the engine inlet.

(4) The fuel level should be the lowest consistent with test safety. The reference to full fuel tanks in § 29.961(c)(2) is misleading because:

(i) Section 29.955(a)(2) would require adequate fuel flow under low fuel level conditions.

(ii) The provision of § 29.961(a)(2) to verify satisfactory hot fuel operation "under critical operating conditions" would mean verification at maximum rate of climb and maximum fuel suction head. The maximum fuel suction head would occur with lowest fuel level.

(5) The flight tests to the service ceiling should include maximum power climbs to selected intermediate altitudes where various maneuvers including the following are performed:

(i) Low power descent with rapid transition to takeoff power.

(ii) Turns and cyclic pull-ups with load factors comparable to the flight strain survey.

(iii) For multiengine rotorcraft with 30-minute and/or 2.5-minute OEI power ratings, conduct a rapid single-engine acceleration from low power to engine topping power followed by cruise at 30 minute OEI power.

(6) The flight test maneuvers should be repeated at the service ceiling.

(7) Except for transients and descents, the power available used should correspond to a 100° F sea level day lapsed 3.6° F/1,000-foot pressure altitude.

(8) Engine operation throughout the test should be normal; i.e., no surge, stall, flameout, etc., and the engine fuel inlet requirements should not be violated.

(9) Alternative tests on appropriate test rigs may be conducted ensuring proper simulation of altitude, ambient temperature, fuel temperature, fuel flow, and load factors.

AC 29.961A. § 29.961 (Amendment 29-26) FUEL SYSTEM HOT WEATHER OPERATION.

a. Explanation. Amendment 29-26 simplifies and restates the fuel system hot weather certification requirements. This eliminates detail requirements in the existing rule which were, to some extent, redundant, or not necessarily critical for some rotorcraft. The phrase, "including, if applicable, the engine operating conditions defined by §§ 29.927(b)(1) and (b)(2)," was added to ensure that certain critical certification aspects are properly considered.

b. Procedures. This paragraph specifies that all suction lift systems and any other fuel system that may be conducive to vapor formation, show satisfactory engine fuel inlet conditions (within criteria established by the engine manufacturer) when using the fuel with the highest true vapor pressure (TVP) at 110° F fuel temperature. Engine operating conditions should include those defined by §§ 29.927(b)(1) and 29.927(b)(2). Compliance can be shown by analysis, testing, or a combination of both.

AC 29.963. § 29.963 FUEL TANKS: GENERAL.

a. Explanation. Section 29.963(a) sets forth general requirements for fuel tank structural aspects. Paragraph (b) requires design features to react forces defined by § 29.561 without leaking fuel. Paragraph (c) requires that whenever flexible fuel tank liners are used, they must be FAA/AUTHORITY approved. Paragraph (d) requires that integral fuel tank interiors be inspectable and repairable.

b. Procedures.

(1) For compliance with § 29.963(a), the tests of § 29.965 are normally adequate if performed in conjunction with the reliability test of § 21.35 or other simulation tests.

(2) For compliance with § 29.963 (b), a structural analysis is usually required to show adequate strength under the loads of § 29.561. Testing, if proposed, may also be an acceptable method of compliance.

(3) For compliance with § 29.963(c), prior FAA/AUTHORITY approvals should be reviewed to ensure compatibility with current project requirements. Also, if a new approval is required as part of the project, then analysis and/or tests should be conducted as appropriate to ensure compliance.

(4) For compliance with § 29.963(d), a review of the design data and/or a visual inspection of any prototype available for inspectability and repairability considerations is usually sufficient to determine compliance. Features such as inspection ports and access panels are typical methods of compliance.

AC 29.963A. § 29.963 (Amendment 29-26) FUEL TANKS: GENERAL.

a. Explanation. Amendment 29-26 adds § 29.963(e) that requires designs and tests to ensure that no exposed surface inside a fuel tank would, under normal or malfunction conditions, constitute an ignition source. It also sets forth standards for the design and qualification of fuel tanks located in personnel compartments. These requirements are needed to ensure freedom from the hazards of fuel tank internal explosions and to ensure that fuel tanks installed in passenger compartments present no hazards to the personnel or to the rotorcraft.

b. Procedures. Section 29.963(e) requires the temperature of any exposed surface inside a fuel tank to be at least 50° F lower than the lowest auto-ignition temperature of the fuel or fuel vapors in the tank (reference paragraph AC 29.1185b(3), § 29.1185). For compliance with § 29.963(e), the internal component surface temperatures can be determined by flight or laboratory tests. The most critical flight conditions are established with sensitive temperature and pressure measuring equipment. This equipment is installed inside the tanks and in the ventilation air spaces.

AC 29.963B. § 29.963 (Amendment 29-35) FUEL TANKS: GENERAL.

a. Explanation. Amendment 29-35 adds a new paragraph (b) that includes the requirements previously contained in paragraph (c) that each flexible fuel tank bladder or liner be either FAA/AUTHORITY approved or be suitable for each particular installation. In addition, the new paragraph (b) adds the requirement that the fuel tank bladder or liner be puncture resistant by meeting the TSO C80, paragraph 16.0, screwdriver test requirements, using a new crash resistance based minimum puncture force of 370 lbs. The requirements previously contained in paragraph (b) are replaced by the crash resistant fuel system requirements of § 29.952 (including load factors). A new paragraph (e) is also added. Paragraph (e) requires that each fuel tank installed in a personnel compartment be isolated by fume-proof and fuel-proof enclosures that are drained and vented to the exterior of the rotorcraft. Further, the design and construction of the enclosures must provide the necessary protection for the tank, must be crash resistant by meeting the applicable criteria of the new Crash Resistant Fuel System requirements of § 29.952, and must be adequate to withstand the loads and abrasions to be expected in personnel compartments.

b. Procedures.

(1) Paragraph (b). The procedures for paragraph (c) prior to Amendment 29-35 still apply to new paragraph (b). In addition, to comply with the added puncture resistance requirement under new paragraph (b), the requirements of § 29.952(g) must be met. Paragraph AC 29.952 gives the detailed compliance procedures for § 29.952(g). The compliance procedures for § 29.952(g) also provide compliance for puncture resistance under § 29.963(b).

(2) Paragraph (e). Compliance with paragraph (e) can be shown by conducting a thorough design review of each fuel tank and its enclosure that is installed in a personnel compartment to ensure the regulatory criteria are met. (All fuel drains and vents should also be reviewed to ensure that they meet applicable § 29.952 requirements.) A basic static loads analysis followed by a stress analysis is typically used to determine that the enclosure protects the fuel tank and provides the crash resistance level necessary for occupant survival in an otherwise survivable impact. The applicable emergency load factors are typically used to design the enclosure. (Section 29.952 contains the corresponding load factors for fuel cells and their attachments.) The emergency load factors are typically adequate for all loading conditions encountered by the enclosure in service. The typical design approach is to design the enclosure to crush at a rate approximately the same as the crush rate of the fuel tank and to ensure that all puncture hazards (such as sharp projections either enhanced or created by impact that would penetrate the fuel tank) are minimized in

design. (See paragraph AC 29.952 guidance material for details.) The design of the enclosure should also be reviewed for overall durability and resistance to all reasonable occupant abuses that could cause a hazard to the integrity of the enclosure, the fuel tank, its vents and its drains.

AC 29.965. § 29.965 (Amendment 29-13) FUEL TANK TESTS.

a. Explanation.

(1) This section prescribes the fuel tank structural tests to be accomplished without failure or leakage.

(2) Section 29.965(b) prescribes pressure testing for conventional metal tanks, integral tanks, and for nonmetallic tanks with walls that are not supported by the rotorcraft structure.

(3) Section 29.965(c) prescribes pressure testing for nonmetallic tanks with walls supported by the rotorcraft structure.

(4) Section 29.965(d) prescribes slosh and vibration testing for tanks with large unsupported or unstiffened flat areas.

b. Pressure Tests.

(1) Each conventional metal tank, integral tank, and each nonmetallic tank without supporting rotorcraft structure should be subjected to pressures of at least 3.5 PSI gage.

(i) If the pressures developed during maximum limit acceleration or emergency deceleration with a full tank exceeds the 3.5 PSI value, a hydrostatic pressure test (or equivalent) should be used to duplicate these acceleration loads as far as possible.

(ii) Pressures need not exceed 3.5 PSI on surfaces not exposed to the acceleration loading.

(iii) Section 29.337 gives the value for the maximum limit acceleration.

(2) Section 29.965(c) applies to nonmetallic tanks with walls supported by the rotorcraft structure. Section 29.965(c)(1) does not require that the tank alone be capable of withstanding 2.0 PSI. Rather, the tank may be mounted in the supporting structure and subjected to the testing of § 29.965(c)(2).

(3) Pressure tests may be conducted by slowly applying a controlled, gauged air source to the tank with sealed vents and fluid entrances and exits. The air pressure

source should then be positively sealed and the tank should retain the prescribed pressure.

(4) Tank and surrounding structure should be carefully examined during and after pressure testing to ensure that there is no damage.

(5) If the prescribed 3.5 PSI or 2.0 PSI, depending on the type of tank, will be exceeded on some surfaces during maximum limit acceleration loading, hydrostatic testing may be preferred. High density fluids have been used to apply the acceleration loads to lower surfaces with supplemental air pressure used above the liquid surface to provide the appropriate pressure on upper surfaces.

(6) For fuel tanks in those areas designated by § 29.967(f), the pressure tests may be designed such that compliance with that paragraph also demonstrates compliance with § 29.965 pressure test requirements.

(c) Slosh and Vibration Tests.

(1) The test requirements of § 29.965(d) are very specific and require little explanation.

(2) There is not an absolute value of what constitutes “large” unsupported or unstiffened flat areas. However, it has generally been considered that any fuel tank with less than a 10-gallon capacity, constructed with a simple, wide, flat geometric shape and using metal (in metal tanks) of 0.05-inch thickness or greater would not require tests in accordance with § 29.965(d). Using this basis, a 14- by 14- inch properly constructed tank would not require vibration and slosh tests.

(3) If the tank construction is of a metal or integral design which can be shown to be similar to previously approved tanks with acceptable service history, the vibration and slosh tests may not be required. Similarity would entail comparing the construction technique; i.e., similar panel size, similar sealing methods, skin and angle thickness, similar loads, etc.

(4) For fuel tanks located in a sponson or stub wing, the entire sponson or wing should be rocked and vibrated unless it can be determined that a certain portion of the tanks is critical. In this case a fixture should be developed such that the portion of the tank being tested is rocked about a pivot point which would produce the same amplitudes of motion for the portion of the tank being tested, as if the whole sponson or wing was being tested. Structural loads in conjunction with these tests have not been required.

(5) The amplitude of vibration specified in the regulation is double amplitude (peak-to-peak). Vibration amplitudes less than one thirty-second of an inch should be justified by instrumented tests of the tank installed in the aircraft.

(6) The vibration and slosh procedures listed in Military Specification MIL-T-6396 have been accepted to show compliance with § 29.965(d).

(7) After all tests have been conducted, the tanks should be leak checked using test fluid conforming to Federal Specification TT-S-735 type III or equivalent.

AC 29.967. <u>§ 29.967 FUEL TANK INSTALLATION</u>.

a. <u>Section 29.967(a)</u>:

(1) <u>Explanation</u>. This paragraph sets forth a series of detail requirements for fuel tanks intended to ensure that tank leakage or failure is unlikely. These requirements pertain primarily to proper support of the tank and protection against chafing.

(2) <u>Procedures</u>. For conventional metal tanks, the support devices, commonly called “cradles,” should be designed with wide flanges or cap strips at the contact area with the tank to distribute the loads in the tank material. To prevent chafing, install nonmetallic padding, treated to eliminate absorption of fuel between the tank and the support structure. Cork strips sealed with shellac and bonded to the support structure have been found suitable. Fuel cell sealant material should be applied over rivet heads and in corners. Bladder cells must be designed to fit accurately in the cell cavity in order to avoid fluid loads in the bladder itself. The interior of the cavity should be smooth to avoid damage to the bladder cells.

b. <u>Section 29.967(b)</u>:

(1) <u>Explanation</u>. This paragraph requires the design to provide ventilation and drainage of spaces adjacent to fuel tanks to avoid accumulation of fuel or fumes to be expected from minor leakage of fuel tanks. This is needed to minimize the possibility of fire or explosion in these spaces. An exception to this requirement is allowed for bladder cells installed in a closed compartment. For this configuration, ventilation may be limited to that provided by compartment drains if the ventilation is adequate to maintain proper pressure relationship between the bladder cell and cell compartment air spaces.

(2) <u>Procedures</u>. With the assumption that fuel tank leakage will occur, require the tank compartments to be provided with drains at any low point. These drains should conduct fuel clear of the rotorcraft and should be three-eighths of an inch or larger in diameter to minimize clogging. As with any drain intended to function in flight, verification that reverse flow will not occur due to pressure differentials at each end of the drain is appropriate. Ventilation for these tanks should involve openings in the compartment walls such that in-flight slipstream and/or rotor downwash will rapidly and continuously purge the tank compartment of fuel fumes. Openings should not be located so the fumes or fuel can reenter the rotorcraft. For flexible tank liner configurations (bladder cells), no specific ventilation is required if the cell is located in a

compartment which is closed, except for drain holes. Note that a cell leak may be expected to produce fumes in the compartment airspace which are flammable; thus, items installed in bladder tank cavities shall not create a hazard during either normal or malfunction conditions. The vent system for the interior of the cell must be adequate to ensure that the bladder cell interior pressure is always positive or at least neutral with respect to any other airspace in the cell compartment to prevent collapse of the bladder cell. Drainage of the cell compartment should meet the criteria discussed above.

(3) A light mesh or string network hung between the bladder cell and its compartment walls is recommended to provide seepage channels to facilitate fuel leakage to the low-point compartment drains.

c. <u>Section 29.967(c)</u>:

(1) <u>Explanation</u>. This paragraph requires a measure of protection for fuel tanks from adverse effects of a fire in a fire zone.

(2) <u>Procedures</u>. Verify that a firewall meeting the requirements of § 29.1191(e) effectively separates any fuel tank from any engine. To minimize hazards of heat transfer to a fuel tank through a firewall during an engine compartment fire, verify that at least one-half inch of clear airspace exists between the tank and the firewall.

d. <u>Section 29.967(d)</u>:

(1) <u>Explanation</u>. This paragraph is intended to prevent hazards to integral fuel tanks to be expected by impingement of flames or products of combustion from an engine compartment fire.

(2) <u>Procedures</u>. Review the design for relative positions of engine compartments and integral fuel tanks to estimate the flowpath of fire or heat from an engine compartment fire. Consider autorotation for single-engine rotorcraft and, for multiengine rotorcraft, low power descent as power-on flight in this evaluation. If questionable compliance exists, clear indication of the flow impingement patterns may be identified by ejecting a dye from engine compartment openings during flight.

e. <u>Section 29.967(e)</u>:

(1) <u>Explanation</u>. This paragraph is primarily intended to provide a standard for installing fuel tanks in personnel compartments. The primary safety concern is to isolate fuel or fumes from personnel in event of a leak in the tank.

(2) <u>Procedures</u>. Assume a leak in the tank and determine that, through the use of additional walls, bulkheads, enclosures, etc., that fuel and fumes will be safely drained and/or purged to the exterior of the rotorcraft. Note that, in order to perform their intended function, the enclosure material and structure should withstand the

mechanical stresses and abrasions to be expected from crew and passenger activities within the compartment.

f. Section 29.967(f):

(1) Explanation. This paragraph is intended to require the design to prevent fuel tank or tank support failure when exposed to the minor crash loads of § 29.561 if such failure could result in fuel entering personnel compartments or fire hazard areas.

(2) Procedures. If a review of the design indicates that tanks are in or adjacent to passenger compartments, or are adjacent to combustion heaters or engines (including APU's), further evaluation of the structural integrity of the tank and its support features must be accomplished. Normally, this involves a quantitative analysis of the tank support structure to confirm that it can sustain the minor crash loads plus one or more pressure tests to simulate the fluid loads on the tank interior to be expected when the minor crash loads are applied. This latter requirement may be, in many cases, satisfied by the qualification requirements of §§ 29.963 and 29.965. Pressure tests tend to overstress upper surfaces of a tank in order to achieve the required stress in the lower surfaces. To minimize this, some applicants have filled the tank to be tested with high density fluids and applied only supplemental pressure to the airspace at the top of the tank. High density fluids are available from the petroleum industry.

AC 29.967A. § 29.967 (Amendment 29-35) FUEL TANK INSTALLATION.

a. Explanation. Amendment 29-35 removes paragraph (e) from § 29.967 and places the identical criteria in a new paragraph (e) to § 29.963. This was done to make §§ 29.963 and 29.967 parallel with §§ 27.963 and 27.967.

b. Procedures. The procedures specified in paragraph AC 29.967, subsection (e) now apply under paragraph AC 29.963B. Thus there is no change in the certification requirements or the compliance methodology, only a change in their location in the FARs and Advisory Material, respectively.

AC 29.969. § 29.969 FUEL TANK EXPANSION SPACE.

a. Explanation.

(1) Space must be provided in each fuel tank system to allow for expansion of the fuel as a result of a fuel temperature increase. The space provided for this purpose must have a minimum volume equal to 2 percent of the tank capacity.

(2) The fuel tank filling provisions must be designed to prevent inadvertent filling of the fuel tank expansion space when fueling the rotorcraft in the normal ground attitude on level ground.

b. Procedures.

(1) Fuel tanks with interconnected vents need not have provisions for fuel expansion in each tank if equivalent expansion provisions are available in another area.

(2) The fuel filler ports should be located below the designated fuel expansion space height to assure that the fuel expansion space cannot be inadvertently filled with fuel. For pressure refueling systems, compliance with this section may be shown with the means provided to comply with § 29.979(b).

(3) Each fuel tank expansion space must comply with the venting requirements of § 29.975.

(4) For multiengine rotorcraft using a single expansion tank to satisfy the requirements of this regulation, the effect of blockage or failure of any vent from this common tank must be considered with respect to compliance with the applicable engine isolation requirements.

AC 29.969A. § 29.969 (Amendment 29-26) FUEL TANK EXPANSION SPACE.

a. Explanation. Amendment 29-26 was issued so that properly interconnected fuel tanks will not be required to have an expansion space for each tank if adequate expansion space is otherwise provided. This amendment eliminates unnecessary design requirements when simpler designs have been proven to be satisfactory.

b. Procedures. Methods of compliance are not changed with this amendment.

AC 29.971. § 29.971 (Amendment 29-12) FUEL TANK SUMP.

a. Explanation.

(1) Each fuel tank should be provided with a drainable sump which is located at the lowest point in the tank with the rotorcraft at ground attitude in order to allow drainage of possibly hazardous accumulations of water from the system.

(2) The minimum required sump capacity, 0.10 percent of the tank capacity or one-sixteenth of a gallon, whichever is greater, should be effective at any normal attitude and located such that the sump contents cannot escape from the tank outlet opening.

(i) Combined interconnected tanks can be treated as a single tank and utilize only one sump if that sump can be located to allow effective trapment and drainage of the potential combined water accumulation.

(ii) The requirement that sump contents not be allowed to escape through the tank outlet opening is intended to ensure that water, or other impurities which may precipitate from the fuel in the tank(s), does not enter the fuel feed system.

(3) Section 29.971(c) would ensure that the fuel tank design and installation allows drainage of hazardous quantities of water to the sump with the rotorcraft in the ground attitude.

(4) Section 29.971(d) would ensure that not only are possibly hazardous accumulations trapped, but also that they are drainable with the rotorcraft in the ground attitude.

(5) Proposed Amendments (Notice 84-19) to §§ 29.971(c) and 29.999(a) would require that the tank sumps be designed or arranged to collect water and be drainable in any ground attitude to be expected in service. This proposed provision would require consideration of the effectiveness of the sumps and drains at the sloped landing limits as well as at normal ground attitude.

b. Procedures.

(1) Demonstration of compliance with the minimum sump capacity requirements may be shown by analysis, test, or a combination of both depending on the complexity of the fuel system design.

(2) If minimum sump capacity is to be established by tests, the following procedure has been accepted.

(i) Fuel the aircraft tanks to ensure that all sumps are filled, that any transfer pumps are immersed, and that the fuel level is above the fuel feed pickup point in the tank(s).

(ii) Use the normal fuel feed provisions to remove fuel from the system. The fuel inlet line at the engine/airframe interface may be disconnected and the fuel pumped overboard. If an engine-supplied suction lift pump is the normal feed mechanism, a suction lift pump of approximately the same capability may be substituted to avoid operating the engine.

(iii) Determine the most critical ground attitude to be expected in service from such considerations as uneven terrain, slope landing limits, etc. The critical attitude for each tank will be that for which the maximum amount of fuel can be withdrawn from the tank using the rotorcraft's fuel supply system.

(iv) Using a rotorcraft with a fuel system which conforms to the final design specification, position the rotorcraft to the critical attitude for the tank to be tested using leveling jacks, actual terrain of a predetermined slope, or other similar means.

(v) Using the rotorcraft's fuel supply system, pump fuel from the tank being tested until the supply system will no longer withdraw fuel. This can be done without the rotorcraft engine actually running unless an engine driven pump is an essential component of the fuel supply system. Caution should be exercised if an engine is to be run to fuel exhaustion since engine surge at the pump cavitation point can result in damaging torsional loads in the transmission drive system.

(vi) When no more fuel can be removed from the tank with the rotorcraft fuel supply system, return the rotorcraft to a normal ground attitude. Completely drain the sump of the tank or tanks being tested into a container and measure the volume drained from each sump. The volume measured must satisfy the minimum capacity requirements of paragraph AC 29.971(a)(2).

(3) If, in the above procedure, a known quantity of fuel is added to initially empty tanks and the total fuel removed (pumped overboard and drained) is recorded, the data may also be used to show compliance with §§ 29.971(d) and 29.999(a).

AC 29.971A. § 29.971 (Amendment 29-26) FUEL TANK SUMP.

a. Explanation. Amendment 29-26 requires that fuel tank sump designs be arranged so that drainage from the sump area will be effective with the rotorcraft parked in any allowable ground attitude in lieu of “normal” attitude as previously required.

b. Procedures. All of the policy material pertaining to this section remains in effect with the extra requirement that each fuel tank should be provided with a drainable sump which is located at the lowest point in the tank with the rotorcraft at “any” ground attitude in order to allow drainage of possibly hazardous accumulations of water from the system. This provision requires consideration of the effectiveness of the sumps and drains at the sloped landing limits as well as at normal ground attitude.

AC 29.973. § 29.973 FUEL TANK FILLER CONNECTION.

a. Explanation.

(1) Fuel tank filler connections must be designed so that no fuel can enter into any part of the rotorcraft other than the fuel tank during fueling operations. Spilled fuel must be considered as well as fuel entering the fuel filler port.

(2) A recessed filler connection that can retain appreciable quantities of fuel should have a drain that discharges clear of the rotorcraft.

(3) Section 29.1557(c)(1) prescribes the marking of the filler.

(4) The filler cap must be fuel-tight under the pressures expected in normal operation.

(5) For Category A rotorcraft, the filler cap or cover must warn if the cap is not fully locked or seated. An improperly locked and seated fuel cap should be evident on the preflight inspections.

(6) The parallel Part 23 and 25 requirements specify that, except for pressure refueling connection points, the filling point must have a provision for electrically bonding the aircraft to ground fueling equipment. Though not specifically required by Part 29, rotorcraft manufacturers have included this provision in recognition that the same potential hazard exists for possible discharge of sparks between the fuel dispensing nozzle and the aircraft as would exist for airplanes.

(7) A proposed rule (Notice 84-19) would add a fuel system lightning protection requirement for rotorcraft. The potential for fuel vapor ignition near the filler cap would be a primary concern. (NASA publication 1008, Lightning Protection of Aircraft, and the user's manual to AC 20–53A, Protection of Aircraft Fuel Systems Against Fuel Vapor Ignition Due to Lightning, provide further information.)

b. Procedures. Compliance with the requirements of this paragraph can normally be demonstrated by analysis and physical inspection of the fuel filler connection design. Testing is not normally required.

AC 29.973A. § 29.973 (Amendment 29-35) FUEL TANK FILLER CONNECTION.

a. Explanation.

(1) Amendment 29-35 revised the requirements for fuel tank filler connections. paragraph (a) is revised to require that all fuel tank filler connections be made crash resistant in accordance with the requirements of § 29.952(f) and its associated advisory material (reference paragraph AC 29.952).

(2) Paragraph (a)(3) is revised to require that all filler caps remain fuel tight under fuel pressures induced during a survivable impact.

(3) Paragraph (b) is revised to require that all transport category rotorcraft (not just Category A as currently required) have a filler cap cover or filler cap that warns when the cap is not fully locked or seated on the filler connection. This change ensures that a loose filler cap will not allow spilled fuel and cause a postcrash fire in an otherwise survivable accident.

b. Procedures.

(1) The compliance procedures for general paragraph (a) are those of § 29.952(f) and those described herein for the three subparagraphs to (a).

(2) The compliance procedures for (a)(1) and (a)(2) can normally be demonstrated by analysis and physical inspection of the fuel filler design. Testing is not normally required.

(3) The compliance procedures for (a)(3) are as follows: The fuel tank filler connection must be shown to be leak free under the worst case fuel pressures (due to combination of static pressure and sloshing induced head) from both normal operations and from a survivable impact. The worst case loads from these two conditions must be determined. In most cases the load resulting from a survivable impact will prevail. For the survivable impact, normally the worst case combined pressure loading occurs at the time of impact at the fuselage that places the filler tube neck (at the vicinity of the filler cap connection) in a vertical or near vertical attitude. Once the critical load case is determined by analysis, test, or a combination; the fuel tank filler connection (or an approved mockup) can be tested for sealing capability by applying a fluid such as water at the critical pressure at the critical attitude of the tube (with the cap inverted) for a period of at least 5 minutes. If no significant leakage occurs, then compliance has been shown. Significant leakage is defined as leakage in excess of 10 drops per minute at any time during or after the 5-minute test.

(4) Compliance procedures for paragraph (b) are as follows: Visual means, such as placards and alignment marks, and mechanical means, such as detents and locking slots, must both be provided. This is necessary to give both a clear visual and mechanical indication that a filler cap or a filler cap cover is properly installed and fuel tight after each removal and replacement. Visual indications such as alignment marks, that show proper installation should be easily read from a distance of at least 5 feet by anyone making a routine inspection or check.

AC 29.975. § 29.975 FUEL TANK VENTS AND CARBURETOR VAPOR VENTS.

a. Explanation. This section sets forth design requirements that address siphoning of fuel, pressure differentials, moisture accumulation, fumes in personnel compartments, and carburetor vapor vents.

b. Procedures. The design of the vent for the fuel system should be adequate to preclude problems associated with this section. Analysis and/or flight testing may be required to demonstrate this adequacy depending upon the fuel system design. If flight testing is required, the following flight test procedure is one method of verifying proper vent system operation.

(1) Using a rotorcraft with a fuel tank and vent system which conforms to production design specifications, install differential pressure instrumentation to measure the difference between the gas pressure inside each fuel tank expansion space and the air pressure in the cavity or area surrounding the outside of the fuel tank.

(2) Conduct ground and flight tests, recording the differential pressures between the inside and the outside of the fuel tanks. The following conditions should be evaluated:

(i) Refueling and defueling (if applicable).

(ii) Level flight to V_{NE}.

(iii) Maximum rate of ascent and descent.

(3) Compare the measured differential pressure values with the maximum allowable for the fuel tank design being evaluated. For flexible, bladder-type fuel cells, the pressure inside the tank should not be significantly less than the surrounding pressure to avoid the possibility of collapsing the bladder.

AC 29.975A. § 29.975 (Amendment 29-26) FUEL TANK VENTS AND CARBURETOR VAPOR VENTS.

a. Explanation. Amendment 29-26 adds § 29.975(a)(7) which requires that fuel tank vent systems be designed to minimize fuel spillage and subsequent fire hazards in the event of rollover of the rotorcraft during landing or ground operations.

b. Procedures. All of the policy material pertaining to this section remains in effect with the added requirement that the fuel tank vent system design should minimize spillage of fuel in the vicinity of a potential ignition source in the event of rollover during landing or ground operation.

AC 29.975B. § 29.975 (Amendment 29-35) FUEL TANK VENTS AND CARBURETOR VAPOR VENTS.

a. Explanation. In addition to the current requirements, Amendment 29-35 revises paragraph (a)(7) to add the requirement that the venting system be designed to minimize fuel spillage through the vents to an ignition source in the event of a fully or partially inverted rotorcraft fuselage attitude following a survivable impact. (A survivable impact is defined in paragraph AC 29.952.) Since rotor action on impact and other impact dynamics have been found in numerous cases to cause rollovers or other unusual postcrash attitudes, compliance with this paragraph would significantly mitigate the postcrash fire hazard by minimizing fuel spills through vents to ignition sources when the postcrash attitude of the rotorcraft would allow gravity and/or post impact sloshing induced fuel spills through a normally open fuel vent.

b. Procedures.

(1) In addition to the compliance procedures for the previous amendment; installation of design features, such as gravity activated shuttle valves in the vent lines (that are normally open but close under certain predictable, postcrash scenarios that

are generated by involvement in a survivable impact that results in either an inverted or partially inverted fuselage attitude) must be accomplished.

(2) Once selected, the design feature chosen for compliance should be shown to function effectively without significant leakage by either full-scale and/or bench tests that apply the total pressure forces that correspond to a 100 percent full, 50 percent full, and 5 percent full fuel load applied to the device in a worst case survivable impact. (If a critical fuel level can be clearly identified, then only that fuel level and the corresponding critical total pressure load need be utilized for certification approval.) The total pressure forces should be determined and applied in a manner that simulates the magnitude and rate of load onset (due to a combination of gravity and sloshing) that would occur in otherwise survivable impacts that would involve rollover attitudes of 45 degrees (or the minimum spillage roll angle), 90 degrees (rotorcraft on its side), and 180 degrees (rotorcraft fully inverted). (In some designs, the 45-degree attitude may not be the correct initial roll angle at which fuel spillage through a given vent would begin to occur due to the placement of the vents on the fuselage. For these cases, the minimum angle should be determined by analysis.)

(3) Once all test conditions are defined, these tests should be conducted with all structural deformation present in the test set up that is necessary to simulate the actual structural deformation either in or applied to the vent line or system in a worst case survivable impact. The structural deformation to be applied can be determined by rational analysis, analysis, test, or a combination. Significant leakage is defined as leakage of 10 drops per minute, or less, after all testing is complete. The criteria of 10 drops per minute, or less, corresponds to the criteria of 5 drops per minute, or less, per breakaway coupling half (i.e., a total of 10 drops per minute, or less, for the entire separated coupling) specified in the advisory material for § 29.952 (reference paragraph AC 29.952).

AC 29.977. § 29.977 (Amendment 29-12) FUEL TANK OUTLET.

a. Explanation.

(1) This section prescribes a fuel strainer for the fuel tank outlet (suction lift system) or for the booster pump (boosted systems) for both reciprocating and turbine engine installations.

(2) This requirement ensures that relatively large, loose objects which may be present in the fuel tank do not interfere with fuel system operation. The provisions of § 29.997 should ensure protection from smaller contaminants which may occur in service.

b. Procedures.

(1) Section 29.977(a) specifies an 8- to 16-mesh-per-inch strainer for reciprocating engine installations, and a strainer which will prevent passage of any

object which could restrict fuel flow or damage any fuel system component for turbine installations.

(2) In addition to the requirement of § 29.977(a), the flow area of the strainer should be at least five times the area of the outlet line. Furthermore, the diameter of the strainer must be at least that of the fuel tank outlet line.

(3) Each finger strainer should be accessible for inspection and cleaning.

(4) Compliance with § 29.977 is usually verified by inspection, and testing is not required. The ice protection provisions of § 29.951(c) are applicable to the strainer at the fuel outlet, and testing to show compliance with that provision may be required.

AC 29.979. § 29.979 (Amendment 29-12) PRESSURE REFUELING AND FUELING PROVISIONS BELOW FUEL LEVEL.

a. Explanation.

(1) Each fueling system that has the fueling connection below the fuel level in the tanks must prevent the loss of fuel if the fuel entry valve malfunctions.

(2) For pressure refueling systems, a back-up limiting device must be provided in addition to the primary means for limiting the amount of fuel in the tank.

(3) Components of the pressure fueling and defueling systems must be able to withstand an ultimate load that is 2.0 times the maximum pressure (positive or negative) most likely to occur during fueling or defueling. This requirement provides a level of structural integrity for the pressure fueling and defueling system components in the event a system malfunction occurs, which would result in an overpressurization of the fuel system. The fuel tanks and vents are not included in this requirement.

b. Procedure.

(1) Designs which have the pressure refueling and fueling provisions below the fuel level in each tank must demonstrate that when there is a malfunction of the fuel entry valve, no hazardous quantity of fuel will be lost. Generally, any amount of fuel loss in excess of 8 ounces is considered to be hazardous. Any amount of fuel that can come in contact with an ignition source is hazardous and unacceptable. Compliance should be demonstrated by test and supported by a failure mode and effects analysis.

(2) For pressure refueling systems, one of the most hazardous failure modes is an undetected overpressurization of the fuel tank which could lead to a number of potential fuel system failures. The pressure refueling system must contain a device which insures that fuel tank capacity cannot be exceeded. This device can operate on a differential pressure principle or can sense fluid level. A back-up limiting device is required in case of failure of the primary limiting device. Compliance must be

demonstrated by test. A failure mode and effects analysis should be performed which verifies that the failure of either the primary or back-up limiting device will not result in the failure of the other limiting device.

(3) The rotorcraft pressure fueling and defueling systems must be designed to withstand an ultimate internal pressure load that is twice the maximum pressure that is likely to occur during fueling or defueling. The maximum pressure will include surges that could occur from the fueling source and/or from any single tank valve or combination of valves being either intentionally or inadvertently closed. System substantiation may be demonstrated by analysis or test. The substantiation should include all components of the pressure fueling and defueling system except the fuel tank and the fuel tank vents. The rotorcraft defueling system must also be substantiated for a negative pressure application. If tests are conducted, the pressure measurements for both tests (positive and negative) will be made at the fueling connection and the test set-up should conform to the installed system.

SUBPART E - POWERPLANT

FUEL SYSTEM COMPONENTS

AC 29.991. § 29.991 FUEL PUMPS.

a. Explanation.

(1) Section 29.991, paragraph (a) provides a definition of the main pump(s) and § 29.991, paragraph (b) requires an "emergency pump(s)." The main pump(s) that is certified as part of the engine does not fall under § 29.991 requirements. The main pump(s) discussed under § 29.991 should therefore be considered "main aircraft pump(s)."

(2) The main aircraft pump(s) consists of whatever pump(s) is required to meet engine or fuel system operation throughout the range of ambient temperature, fuel temperature, fuel pressure, altitude, and fuel types intended for the rotorcraft. If the main aircraft pump(s) is required to meet the above criteria, then an emergency pump(s) is required.

b. Procedures.

(1) Each pump classified as a main aircraft pump, which is also a positive displacement pump, must have provisions for a fuel bypass. An exception is made for fuel injection pumps used on certain reciprocating engines and for the positive displacement, high pressure, fuel pumps routinely used in turbine engines. The bypass may be accomplished via internal spring check valve and fuel passage, or by external plumbing and a check valve. High capacity positive displacement pumps with internal pressure relief and recirculation passages should be checked for overheating if they may be expected to operate continuously at or near 100 percent recirculation.

(2) Section 29.991, paragraph (b) specifies a requirement for "emergency" pumps to provide the necessary fuel after failure of any (one) main aircraft pump. (Injection pumps and high pressure pumps used on turbine engines are exempt.) As stated in this rule, the "emergency" pump must be operated continuously or started automatically to assure continued normal operation of the engine. For some multiengine rotorcraft, another main aircraft pump may possibly be used as the required "emergency" pump. In this case, the dual role of this pump requires it to have capacity to feed two engines at the critical pressure/flow condition. Availability of fuel flow from this backup pump must be automatic and this function should be verified in the preflight check procedure. For Category A rotorcraft, a comprehensive fault analysis of the fuel system is mandatory to assure compliance with § 29.903, paragraph (b).

(3) Section 29.991, paragraphs (c)(1)(i) and (ii) address the situation, usually associated with supercharged reciprocating engines, where fuel pressure must be modulated with respect to carburetor deck pressure. This is accomplished with interconnecting air lines from the carburetor intake (after the supercharger) to the pressure relief connection on the fuel pump(s). A similar connection from the carburetor intake to the vented side of the fuel pressure gauge is needed to obtain correct fuel pressure reading. These systems may require orifices and/or surge chambers to operate correctly.

(4) Section 29.991, paragraphs (c)(2) and (3) requires seal drains which drain safely. A drain impingement test is normally required to verify safe drainage. Use of a colored dye to simulate fuel discharge at the drain line exit or a fluid sensitive coating (Bon Ami) on the aircraft skins will facilitate evaluation of the safety aspects of drain impingement. Pump seal drain requirements would not be applicable for tank immersed pumps.

AC 29.991A. § 29.991 (Amendment 29-26) FUEL PUMPS.

a. Explanation. Amendment 29-26 revises § 29.991 to clarify fuel pump redundancy requirements. Redundancy for fuel pump failure includes consideration of both the pump and the pump motivating device.

(1) Section 29.991(a)(1) now stipulates that a single fuel pump failure should not jeopardize the capability of the fuel system from delivering the fuel necessary to satisfy the requirements of § 29.955. This stipulation excludes any fuel pumps that are approved as a part of the type certificated engine.

(2) Section 29.991(a)(2) expands the stipulation of § 29.991(a)(1) by including any component(s) required to drive the fuel pump (such as electric motors or generators for electric pumps). This section also stipulates that if the pump is engine driven, failure should not affect more than the engine served by the pump.

b. Procedures. The method of compliance for this section is unchanged.

AC 29.993. § 29.993 FUEL SYSTEM LINES AND FITTINGS.

a. Explanation. This rule outlines design requirements for fuel system lines.

b. Procedures.

(1) Compliance is usually obtained by employing routing and clamping as described in paragraph 709, Chapter 14, Section 2, of AC 43.13-1A and by monitoring the arrangement throughout the developmental and certification test period. Requirements for approved flexible lines may be resolved by utilizing lines listed as TSO C53a approved for installation in either normal or high temperature areas as appropriate.

(2) Verify adequate clearance exists between lines and elements of the rotorcraft control system at extremes of control travel, including control deflections and, for flexible lines (hoses), possible variations in routing.

(3) Flexible lines inside fuel or oil tanks require special evaluation to assure that the external surfaces of these lines are compatible with the fluids involved and that fluid sloshing will not cause line failure. Lines inside tanks should be routed to avoid impingement by fuel or oil filler nozzles.

(4) Good design practice suggests that all flammable fluid lines should be routed to minimize the possibility of rupture in the event of a crash or from engine rotor disc failure.

AC 29.995. § 29.995 (Amendment 29-13) FUEL VALVES.

a. Explanation. This regulation requires that fuel valves be supported so that no loads resulting from their operation or from accelerated flight conditions are transmitted to the lines attached to the valve.

b. Procedures. Compliance with this rule is usually accomplished by designing the installation of the fuel valve so that the valve is supported by either primary or secondary airframe structure.

AC 29.997. § 29.997 (Amendment 29-10) FUEL STRAINER OR FILTER.

a. Explanation. This rule provides for a main in-line fuel filter designed to collect all fuel impurities which could adversely affect fuel system and engine components downstream of the filter. The rule also requires a sediment bowl and drain (or that the bowl be removable for drain purposes) to facilitate separation of contaminations, both solid and liquid, from the fuel.

b. Procedures.

(1) The filter should be mounted in a horizontal segment of the fuel line to facilitate proper action of the sediment bowl. If the filter is located above the fuel tank, it becomes necessary to activate a fuel boost pump to achieve positive drainage of the filter bowl. Without pump pressure, air may enter the fuel system during the filter draining operation and, for turbine engines, result in transient power surges or engine failure during subsequent engine operation. A flight manual note to require pump(s) to be “on” during filter draining would be appropriate.

(2) Section 29.997(d) sets forth a requirement for filter capacity and for filter mesh. The capacity requirement may be substantiated by showing that the filter, when partially blocked by fuel contaminates (to a degree corresponding to the indicator marking or setting required by § 29.1305(a)(17)), does not impair the ability of the fuel

system to deliver fuel at pressure and flow values established as minimum limitations for the engine. The filter mesh must be sized to prevent passage of particulate which cannot be tolerated by the engine. FAR Part 33 requires that the degree and type of filtration be established. This information should be the base for selecting the filter mesh. Although a test may be devised and conducted, data from the filter manufacturer usually are acceptable to verify compliance. Note that when the filter capacity is reached, continued flow of contaminated fuel may result in engine failure. A flight manual note regarding precautionary procedures is appropriate.

(3) FAR Part 33 (through Amendment 33-6) has an identical requirement for a fuel filter for engine fuel systems; however, it is not intended that two filters should be required.

AC 29.997A. § 29.997 (Amendment 29-26) FUEL STRAINER OR FILTER.

a. Explanation. Amendment 29-26 requires that a fuel strainer or filter should be installed between the fuel tank outlet and the first fuel system component that is susceptible to fuel contamination. Components that will be protected from contamination include but are not limited to fuel metering devices which control flow rate, fuel heaters, and positive displacement pumps. The amendment also requires a sediment bowl and drain (unless the bowl is readily removable for drain purposes) to facilitate separation of solid and liquid contaminants from the fuel.

b. Procedures.

(1) The fuel strainer or filter should be accessible for draining and cleaning. It should incorporate a screen or other element that is easily removable. It should be mounted so that its weight is not supported by the inlet or outlet connections of the strainer itself, unless it can be shown that adequate strength margins exist in the lines and connections.

(2) The fuel strainer or filter should have a sediment trap and drain (unless the trap is readily removable for drain purposes). The volume capacity of the sediment trap is specified in § 29.971(a) (0.10 percent of the tank capacity or 1/16 of a gallon).

(3) The fuel strainer or filter mesh should provide the filtration stipulated in the FAA/AUTHORITY-approved engine installation manual that is prepared for the type certificated engine (FAR Part 33).

(4) The fuel strainer or filter should have the capability to remove any contaminant that would jeopardize the flow of fuel that is necessary to meet the requirements of § 29.955. In addition, the strainer or filter should have a bypass system with an impending bypass indicator (Refer to § 29.1305(a)(17)). When the strainer or filter is partially blocked with contaminants, to the degree that the fuel flow requirements of § 29.955 can no longer be achieved, the impending bypass indicator should be activated. At this point, the strainer or filter should not yet be bypassing

unfiltered fuel. Although a test may be devised and conducted, data from the filter manufacturer usually are acceptable to verify compliance. Note that when the filter capacity is reached, continued flow of contaminated fuel may result in engine failure. A flight manual note regarding precautionary procedures is appropriate.

(5) Section 33.67(b) has an identical requirement for a fuel filter for engine fuel systems; however, it is not intended that two filters should be required.

AC 29.999. <u>§ 29.999 (Amendment 29-12) FUEL SYSTEM DRAINS</u>.

a. <u>Explanation</u>. This regulation provides for fuel system drains and defines the requirements which the system must meet.

b. <u>Procedures</u>.

(1) The location and function of the fuel system drains are an integral part of any fuel system. There may be several drains required dependent upon the fuel system design. Each fuel tank sump and certain types of fuel strainers or filters require a means to drain (reference §§ 29.971 and 29.997).

(2) Selection of the location and orientation of the drain discharge in the design phase is important to assure that there is no impingement on any part of the rotorcraft. To show compliance with the requirement may require tests dependent upon whether the applicant has a previously approved design which is similar, or if the system is a new design for which no previous experience is available.

(3) The location of the drain valve should be selected so that the requirements for accessibility, ease of operation, and protection are met.

(4) Advisory Circular 20-119 provides an acceptable means, but not the only means, of compliance with the requirement for positive locking of fuel drain valves in the closed position.

(5) The fuel drain installation on aircraft with retractable landing gear will be satisfactory if recessed within the outside surface of the aircraft.

AC 29.999A. § <u>29.999 (Amendment 29-26) FUEL SYSTEM DRAINS</u>.

a. <u>Explanation</u>. Amendment 29-26 adds the requirement that fuel system drains be effective with the rotorcraft in any allowable ground attitude including uneven terrain. In addition, the change amended § 29.999(b)(2) to require fuel drains have a means to ensure positive closure as contrasted to positive locking when in the “off” position. This will accommodate designs featuring spring-loaded drain closures that have been found to be satisfactory.

b. Procedures. All of the policy material pertaining to this section remains in effect. Additionally, selection of the location and orientation of the fuel drain discharge in the design phase is important to assure that there is no impingement upon any part of the rotorcraft. The location and orientation should also ensure effective fuel drainage when the rotorcraft is parked on uneven terrain. To show compliance with the requirement, tests may be required, dependent upon whether the applicant has a previously approved design that is similar, or the system is a new design for which no previous experience is available.

AC 29.1001. § 29.1001 (Amendment 29-26) FUEL JETTISONING.

a. Explanation. Amendment 29-26 adds § 29.1001 to set forth the certification requirements for a fuel jettisoning system if it is installed in the rotorcraft.

b. Procedures. In showing compliance with the requirements of § 29.1001, the following guidance is provided.

(1) The fuel jettison system should be demonstrated to be safe in all normal flight regimes. Takeoff, hover, and in-ground-effect maneuvers may be excluded if appropriate limitations are prescribed.

(2) The fuel jettison system, and its operation, should be shown to be free from fire hazard. If possible, the fuel should discharge clear of any part of the rotorcraft; however, it should be shown that any fumes or fuel, that do impinge upon the rotorcraft in the form of a fine mist, does not form droplets that run along the exterior structure and enter any part of the rotorcraft (wheel wells, cargo area, tail boom, etc.). It should also be shown that jettisoned fuel is not ingested by the engines or the auxiliary power unit (APU). This demonstration can be conducted by jettisoning a glycol based, dye colored fluid and noting the pattern displayed on a dye sensitive coating applied to the rotorcraft exterior. The demonstration should be conducted over all flight regimes in which system operation is permitted. The demonstration should also take into account the maximum rate of descent and all airspeeds where fuel impingement upon the fuselage would most likely occur. Rotorcraft controllability should remain satisfactory throughout the fuel jettisoning operation and should also be demonstrated.

(3) The requirements in § 29.1001(c) were established to prevent complete fuel depletion and provide the capability to effect continued safe flight and landing.

(4) The controls for the fuel jettison system should be designed so that a “minimum” flight crew can perform the jettison operation and be able at any time to stop the jettison process or begin it again. These design requirements give the flight crew the capability and flexibility to manage their on-board resources.

(5) The requirements of § 29.901(c) are intended to emphasize that no single failure or malfunction or probable combination of failures of the fuel jettisoning system will jeopardize the safe operation of the rotorcraft.

(6) If the rotorcraft has an auxiliary fuel tank, an auxiliary fuel jettisoning system may be installed to jettison the additional fuel provided the jettisoning system has separate and independent controls and it also meets all of the requirements of this section.

SUBPART E - POWERPLANT

OIL SYSTEM.

AC 29.1011. § 29.1011 OIL SYSTEM - GENERAL.

a. Explanation.

(1) The oil system provided for each installed engine should provide all of the lubrication required by the engine and supply it at a temperature which is within the operating temperature limits established for that engine when it was certified.

(2) The usable oil capacity of each oil system should be sufficient to provide oil to the engine at the maximum oil consumption limit of the engine under critical operating conditions. All circulating requirements and operating temperature limits for the oil should be met.

b. Procedures.

(1) There are three basic engine oil supply and cooling system concepts that are used. There are self-contained systems (a complete system certified with the engine), systems that have both engine and airframe components, and systems that are totally supported by airframe components. Any one of these three concepts can be used to meet the requirement of having an independent oil system for each engine.

(2) Oil tank capacity is primarily determined by the engine's oil consumption rate. Other factors which should be considered when sizing the oil supply system are the endurance of the rotorcraft under critical operating conditions, and the amount of oil circulating in the system to maintain proper cooling. Adequacy of the engine oil supply system can be shown by analysis supported by engine oil consumption and cooling system data. For reciprocating engines, the ratio of one gallon of oil for each 40 gallons of fuel can be used; however, an oil-fuel ratio lower than 1:40 can be used if properly substantiated by oil consumption data on the engine.

(3) The engine oil cooling requirements are defined in §§ 29.1041 through 29.1049. The design of the engine oil cooling system will be influenced by hot day conditions, by the engine heat rejection rate, and other oil system operating data provided by the engine manufacturer. Sizing of the oil cooler will depend upon the engine data and whether the oil cooler will also be used for main transmission oil cooling. Oil cooler size should be kept as small as possible due to its effect on rotorcraft structure, but in all cases, adequate cooling should be demonstrated throughout the operating envelope of the rotorcraft.

AC 29.1013. § 29.1013 (Amendment 29-10) OIL TANKS.

a. Explanation. This regulation identifies the requirements that each oil tank must meet. It also specifies that the oil tank installation must meet the installation requirements of § 29.967.

b. Procedures.

(1) The oil tanks usually are constructed of aluminum, aluminum alloy, or stainless steel and are of such a design to permit installation in the aircraft as close to the engine as the design allows. The choice of materials will generally be determined by the selected location of the tank. The tank envelope or outline will generally be determined by the location within the structure of the rotorcraft.

(2) The design of the tank is required to meet the expansion space requirements as specified in the regulation for the particular installation. This is generally accomplished by locating the filler cap in such a manner that the expansion space cannot be inadvertently filled with the rotorcraft in normal ground attitude.

(3) The tank is required to be properly vented and the vent requirements are identified in the regulation.

(4) Unless alternate means are provided, it is good design practice to locate the oil tank with respect to the engine so that when the rotorcraft is in its normal ground attitude, a positive head to the oil pump inlet is provided.

(5) Sections of the regulation address specific requirements when Category A certification is requested.

(6) The designer should be aware of the requirements associated with the location of the oil tank outlet and the marking requirements specified in § 29.1557(c)(2).

(7) Flexible oil tank liners may be used; however, they must be approved or shown to be suitable for the particular installation.

(8) An “external oil system” which is defined as being those components, lines, etc., of an oil system which are outside the engine and not supplied as part of a certificated engine. The components of such a system which are within the fire zone and required to be fire resistant. Those outside the fire zone need not be fire resistant.

AC 29.1015. § 29.1015 (Amendment 29-10) OIL TANK TESTS.

a. Explanation. This regulation defines the tests that must be accomplished to show compliance for rotorcraft oil tanks.

(1) The oil tank should be designed and installed so that it can withstand, without failure, any vibration, inertia, and fluid loads to which it may be subjected in operation.

(2) The installation should meet the requirements of § 29.965 except that for pressurized tanks used with turbine engines, the test pressure may not be less than 5 PSI plus the maximum operating pressure of the tank. For all other tanks, the test pressure may not be less than 5 PSI.

b. Procedures. The pressure tests require that 5 PSI plus operating pressure but in any case no less than 5 PSI be used to substantiate the oil tank. To accomplish these tests, the various tank openings are sealed. An adapter fitting is fabricated by which regulated, pressurized air is introduced into the tank. This air pressure is measured by means of a calibrated air pressure gauge. Any of several methods to determine whether the tank is leaking may be used. As an example, if the tank is relatively small, emergence in a tank of water may be used. Other means such as applying soapy water to the joints are also satisfactory. In any respect, the leak check using test fluid conforming to Federal Specification TT-S-735, Type III, may also be used.

AC 29.1017. § 29.1017 OIL LINES AND FITTINGS.

a. Explanation. This regulation outlines the certification requirements for oil lines and fittings.

b. Procedures. The oil system lines and fittings are required to meet the requirements of § 29.993; therefore, the routing and clamping described in paragraph 709, Chapter 14, Section 2, of AC 43.13-1A may be utilized as guidance for the system design. An evaluation carried out through the development and certification test period will usually surface any problems of interference and/or vibration.

(1) When flexible hoses are used in the lubrication system they must be substantiated. Hoses listed in TSO C53a may be used which would preclude certain substantiation requirements.

(2) Location of the breather lines and discharge should be carefully evaluated to determine that the requirements of this paragraph are followed.

(3) The routing of fluid lines should be such that drooping lines and fluid traps which are undrainable are avoided.

AC 29.1019. § 29.1019 (Amendment 29-10) OIL STRAINER OR FILTER.

a. Explanation. This regulation defines the requirements for the engine oil system strainer or filter. If a strainer or filter which meets the requirements of this paragraph is incorporated as part of the type certificated engine, an additional airframe filter is not required.

b. Procedures. This paragraph requires an oil strainer or filter through which all of the oil flows for each turbine engine installation. The strainer or filter should be sized to allow oil flow at the flow rates and within the pressure limits as specified in the engine requirements. The effect of oil at the minimum temperature for which certification is sought should be accounted for.

(1) For each oil strainer or filter required by § 29.1019(a) which has a bypass, the bypass should be sized to allow oil flow at the normal rate through the oil system with the filtration means completely blocked.

(2) For each oil strainer or filter installed per this rule, the capacity must be such that the oil flow and pressure are within the operating limits established for the engine. The mesh requirements are determined by the engine specification for the filtration of particle size and density.

(3) Section 29.1019(a)(3) requires an indicator that will show when the contaminant level of the filtration system, as specified in § 29.1019(a)(2), has been reached. The indicator should signal a contaminant level which has not caused the filter to go into a bypass condition. Consideration should also be given so that the contaminant level at which the indicator is activated is such that the filter would not bypass during a flight time based on full fuel at a cruise condition with the lubricant contaminated to the degree used to show compliance with § 29.1019(a)(2).

(4) An evaluation of the construction and location of the bypass associated with the strainer or filter should be accomplished. The appropriate installation of the filter based on this evaluation would preclude the release of the collected contaminants in the bypass oil flow.

(5) If an oil strainer or filter installed in compliance with this regulation does not have a bypass, there must be a means to connect it to the warning system required in § 29.1305(a)(18). This warning should indicate to the pilot the contamination before it reaches the capacity established in § 29.1019(a)(2). Section 29.1019(b) covers the blocked oil filter requirements associated with reciprocating engine installations. The lubrication system should be such that the normal oil flow will occur with the filter completely blocked.

AC 29.1019A. § 29.1019 (Amendment 29-26) OIL STRAINER OR FILTER.

a. Explanation. Amendment 29-26 relaxes the requirements of § 29.1019(a)(3) from requiring an indicator to indicate the contamination level of oil filters. The rule change allows acceptance of a “means to indicate” the contaminate level to allow a wider range of acceptable methods of compliance.

b. Procedures. Unless the filter is located at the oil tank outlet, § 29.1019(a)(3) requires that the oil strainer or filter have the means to indicate when the contaminant level of the filtration system, as specified in § 29.1019(a)(2), has been reached. If the indicator is installed, it should signal a contaminant level that will allow completion of the flight before the filter reaches a bypass condition. The indicator may be a pop-out button or other maintenance cue that is checked on each preflight inspection.

AC 29.1021. § 29.1021 OIL SYSTEM DRAINS.

a. Explanation. This regulation requires provisions be provided for safe drainage of the entire oil systems and defines certain requirements for assuring that no inadvertent oil flow occurs from the system provided.

b. Procedures. The design of the oil system must provide a means for safe drainage of the entire oil system. This may require one or more drains dependent upon the design of the system. If a valve is used for this function, it must provide a means for a positive lock in the closed position. The method by which the lock is accomplished may be manual or automatic.

AC 29.1023. § 29.1023 OIL RADIATORS.

a. Explanation. This regulation defines the installation requirements to be considered for oil system radiators.

b. Procedures.

(1) The primary concern with respect to oil radiators is that they are sized to provide the required heat rejection and to provide adequate fluid flow within the prescribed pressure limits.

(2) The structural design of the radiator must consider the system oil pressure requirements and the service involvement of the intended application. The selection of the location of the radiator can have a significant bearing on its ability to withstand the vibration and inertia loads.

(3) If the system design incorporates an air duct to direct the airflow, the effects of a fire as defined in this regulation must be considered.

AC 29.1025. § 29.1025 OIL VALVES.

a. Explanation. This regulation identifies the requirements which oil system valves must meet. In addition to the items specified in this rule, this regulation specifies compliance with the requirements of § 29.1189.

b. Procedures. The closing of the oil shutoffs may not preclude a safe autorotation. Compliance with this requirement is best accomplished in the design phase. This can be accommodated by proper orientation of the valve and/or system plumbing routing. Another means is to design adequate entrapment of lubricants to provide for the autorotation state. The design of the oil shutoff valve must consider the stop or index provisions of this rule. The installation must be such that the loads specified in the rule are addressed.

AC 29.1027. § 29.1027 (Amendment 29-26) TRANSMISSION AND GEARBOXES: GENERAL.

a. Explanation. Amendment 29-26 adds a new § 29.1027. This new section provides the regulations for rotorcraft transmission and gearbox lubrication systems. It incorporates lubrication system requirements that were removed from § 29.1011 and adds additional lubrication system requirements that were derived from existing engine-oil system requirements. These additional requirements have been adjusted or modified to reflect the needs of transmissions and gearboxes. Transmission and gearbox lubrication system regulations are similar to those for engines; therefore, reference is made to the engine lubrication sections as applicable.

b. Procedures.

(1) The pressure lubrication systems for rotorcraft transmissions and gearboxes should comply with the same requirements as the engine lubrication systems stipulated in § 29.1013 (except §§ 29.1013(b)(1), 29.1015, 29.1017, 29.1021, and 29.1337(d)). These sections provide the requirements for oil tanks, tank tests, oil lines and fittings, and oil system drains.

(2) Each pressure lubrication system for rotorcraft transmissions and gearboxes should have an oil strainer or filter. The strainer or filter should:

(i) Remove any contaminants from the lubricant that may damage the transmission, gearbox, or other drive system component and any contaminants that may impede the lubricant flow to a hazardous degree.

(ii) Be equipped with a means to indicate that the bypass system (required by § 29.1027(b)) is at the point of opening, due to the collection of contaminants on the strainer or filter; and,

(iii) Be equipped with a bypass system that will permit lubricant to continue to flow at the normal rate if the strainer or filter is completely blocked. In addition, the bypass system should be designed so that contaminants, that have collected on the filter, will not enter the bypass flow path when the system is in the bypass mode.

(3) Section 29.1027(b)(2) requires a screen at the outlet of each lubricant tank or sump that supplies lubrication to rotor drive systems and rotor drive system components. The screen should remove any object that might obstruct the flow of

lubricant to the filter required by § 29.1027(b)(1). The requirements of § 29.1027(b)(1) do not apply to the tank outlet screen.

(4) Splash-type lubrication systems for rotor drive system gearboxes should comply with §§ 29.1021 and 29.1337(d).

SUBPART E - POWERPLANT

COOLING

AC 29.1041. § 29.1041 COOLING - GENERAL.

a. Background.

(1) Few substantive changes have been made to the cooling provision requirements, §§ 29.1041 through 29.1049, since the rules were defined in the Civil Air Regulations, Part 7, effective August 1, 1956. Testing procedures utilized have not precisely followed those rigorously set forth in §§ 29.1045 through 29.1049 as industry and the FAA/AUTHORITY have recognized the need to vary procedures slightly to accomplish the practical test objectives.

(2) In the paragraphs which follow, the cooling regulations will be explained, and in some instances where the regulations provide specific procedures, "alternative procedures" which have been found acceptable in achieving the rule objectives will be presented. The intent of providing those alternative procedures is not to promulgate new regulations, but rather to provide recognized, accepted procedures for compliance with the objective of the current standards.

b. Explanation.

(1) The rotorcraft design should provide for cooling to maintain the temperatures of all powerplant, auxiliary power unit, and power transmission components and fluids within the limitations established for these items.

(2) Cooling provisions should be adequate for shutdown and for water, ground, and flight operating conditions.

(3) The adequacy of the cooling provisions should be demonstrated by flight testing.

c. Procedures.

(1) Detailed procedures for the demonstration of climb, takeoff and climb, and hover cooling are given in §§ 29.1045 through 29.1049. Other test conditions and procedures necessary to demonstrate adequate cooling for water, ground, flight, and shutdown conditions must be negotiated between the applicant and the FAA/AUTHORITY certification engineer. A cooling test proposal which defines the agreed test points and procedures should be prepared well in advance of the official certification testing.

(2) The test conditions selected, in addition to those in §§ 29.1045 through 29.1049, would typically include cruise at various airspeeds and altitudes, shutdown

after prolonged hover, and sling load cooling if applicable. One test condition which should be examined, particularly with regard to transmission cooling, is the point of highest multiengine mechanical power at the maximum ambient temperature. This is identified as test point “A” in figure AC 29.1041-1. The selection of test points should be tempered with engineering judgment and based on results from similar aircraft, if such data are available.

(3) In showing compliance with the cooling requirements, the applicant should not be required to exceed rotorcraft established limits (gross weight, drive system torque, measured gas temperature, etc.), aircraft power required, or power available. The applicant may elect, however, to exceed these limits in order to minimize test points by conservative testing, or to anticipate future growth (increased gross weight, etc.).

(4) The need for a comprehensive cooling test plan prior to certification testing cannot be overemphasized. Highly derated engine installations, the relationship of power required to power available, the use of bleed air devices which would increase the measured gas temperature while aircraft power required remains the same, auxiliary cooling provisions, and the increase in engine temperatures with engine deterioration are factors which could affect the selection of cooling demonstration test points. The following paragraphs will provide some general guidance, but the cooling test plan is the key to a successful program.

MULTIENGINE POWER AVAILABLE

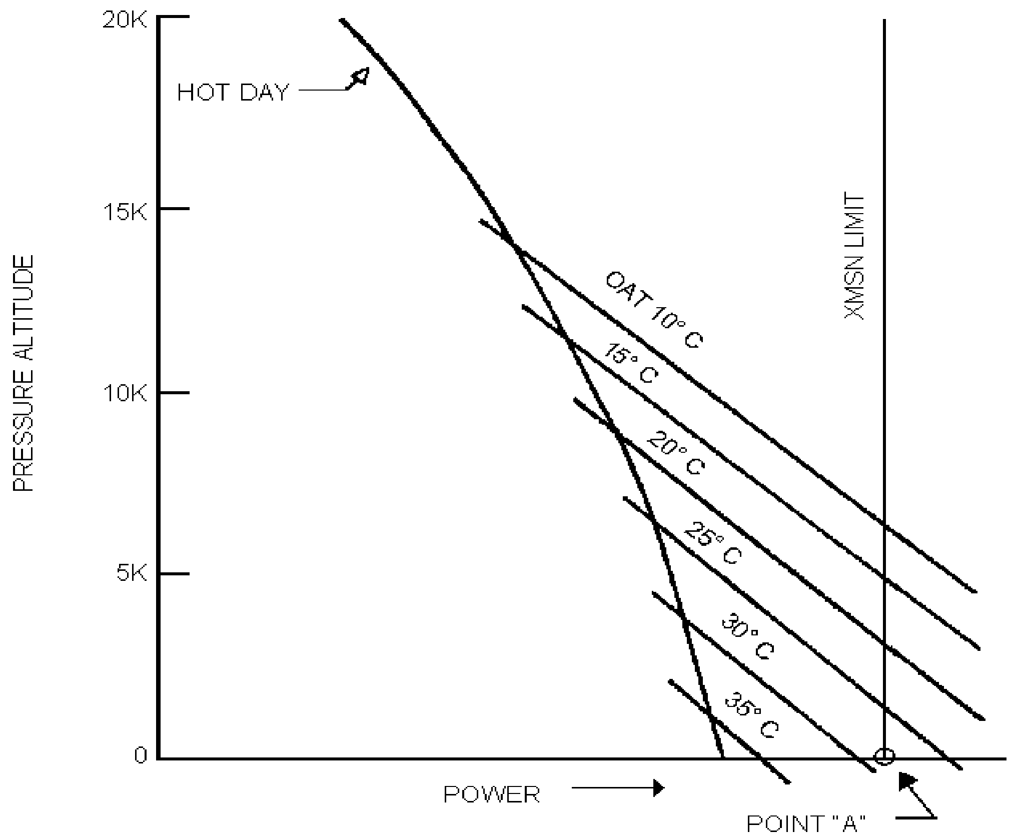


FIGURE AC 29.1041-1 ADDITIONAL COOLING TEST POINT

AC 29.1043. § 29.1043 (Amendment 29-15) COOLING TESTS.

a. Explanation.

(1) Section 29.1043(a) requires that certain ambient temperature correction factors be applied unless testing is accomplished at the maximum ambient atmospheric temperature prescribed.

(2) No corrected temperatures may exceed established limits.

(3) The statement of § 29.1043(a)(4) which requires that test procedures be in accordance with §§ 29.1045 through 29.1049 does not limit testing to the conditions prescribed in those sections. Section 29.1041(a) and (b) provide the basis for examination of other operating and shutdown conditions.

(4) The maximum ambient atmospheric temperature must be at least 100° F at sea level, lapsed to altitude at a rate of 3.6° F per 1,000 feet pressure altitude. The applicant may select a lower maximum ambient atmospheric temperature for winterization installations.

(5) Unless a more rational correction applies, the temperature data (except for cylinder barrels) are to be corrected by adding the difference between the maximum ambient atmospheric temperature and the temperature of the ambient air at the time of the first occurrence of the maximum component or fluid temperature recorded during the cooling test.

(6) Cylinder barrel temperature data are corrected in a similar manner to other components except 0.7 times the difference between the maximum ambient atmospheric temperature and the ambient temperature at the first occurrence of the maximum cylinder barrel temperature is applied.

b. Procedures.

(1) Seldom is testing actually accomplished at the maximum required ambient temperature of at least 100° F at sea level lapsed 3.6° F per 1,000 feet pressure altitude. Component and fluid temperatures must therefore be corrected to derive the item temperature that would have been reached if the test day had matched exactly the maximum ambient temperature day. The applicant may select a higher maximum ambient temperature for cooling certification than the 100° F sea level hot day prescribed. Provisions are also made for selecting a maximum ambient temperature less than the 100° F sea level hot day for winterization installations not intended to function at the hot day conditions.

(2) When cooling test ambient conditions are cooler than the selected or prescribed hot day conditions, the applicant may take advantage of cooling air or fluid

flows that would exist at hot day conditions. For example, thermostatically controlled oil cooler flow could be set for hot day conditions.

(3) The component and fluid temperature correction factor to be applied when test ambients do not correspond to the hot day conditions is commonly called the "degree-for-degree correction." It may be possible to justify, and the regulation allows the application of a more rational, less conservative correction factor. A correction factor other than degree-for-degree should be based on engineering test data.

(4) No corrected temperatures may exceed established limits. In order to maintain temperatures within established limits, the applicant may be willing to accept lesser performance than the full capability of a device. For example, a starter/generator capable of cooling under test cell conditions to 200 amperes continuous load may be limited to a lesser value, perhaps to 150 amperes, when installed in the aircraft due to cooling considerations. This continuous load for cooling must be equal to or greater than the allowable continuous load designated on aircraft instruments.

c. Thermal Limit Correction.

(1) An important correction factor which is not discussed in the regulations, but is frequently necessary to show the cooling adequacy required by § 29.1041, is the thermal limit correction factor. This factor is sometimes required if, at test day conditions, the engine measured gas temperature does not correspond to that which would have occurred on a minimum specification engine at hot day conditions.

(2) The correction factor would not apply to those components not affected by changes in measured gas temperature (MGT) at a constant power. Typical items expected to be affected by changes in the MGT at constant power would be engine oil temperature, thermocouple harnesses, or other fluid, component, or ambient temperatures in the vicinity of the engine hot-section or exhaust gases. Other items remote from the hot-section, perhaps the starter-generator or fuel control, would not be expected to be influenced by MGT variations; however, the items affected and the magnitude of the factor to be applied should be established by testing.

(3) There are several acceptable methods for establishing the appropriate thermal limit correction factor during development testing. The general idea is to establish a stabilized flight condition, typically ground-run or IGE hover, and to vary the measured gas temperature at approximately fixed power and OAT conditions. This may be accomplished by utilizing engine anti-ice bleed air, customer bleed air, or by ingesting warmer than ambient air (either an external source or the engine bleed air) into the engine inlet. Care should be used in ingesting warmer than ambient air to assure that the warm air is diffused in order to avoid possible engine surge.

(i) If it is not possible to attain a suitable variation in MGT by these methods, an acceptable, but more conservative thermal limit correction may be obtained by allowing both shaft horsepower and MGT to vary at a stabilized flight condition and OAT.

(ii) The component temperature is plotted as a function of MGT, and the thermal limit correction from any test day MGT for any flight condition, to the MGT that would have existed with minimum specification engines on a hot day, is then applied to derive the final measured component temperature.

(4) In certain rare instances, it may not be required that the correction factor be applied to the full thermal limit capability of the engine. Consider the following example for the hot day hover IGE cooling test point at sea level.

	Power (SHP)	Corresponding MGT (°C)
Drive System Limit	900	---
Twin-Engine Hot Day Power Available	1,050	750
Hot Day Power Required at Maximum G.W.	850	650
Engine Maximum Allowable MGT (Instrument Marking)	---	765
Test Day (90° F OAT) Parameters	850	600

(i) Notice that the installed hot day power available MGT from the engine performance program, is 15° C cooler than the limit MGT (750° vs. 765° C), thus the engine has 15° C "field margin" which would allow the engine temperature to gradually increase 15° C to maintain a given power as engine life is utilized. Secondly, the measured gas temperature corresponding to hot day power required at maximum gross weight, is less than that corresponding to either the drive system limit or twin-engine hot day power available. Thus, the thermal limit correction could be applied from the test day MGT, 600° C, to the power required MGT plus the field margin, 650° C plus 15° C, rather than applying the correction factor to the full thermal capability of the engine, 765° C.

(ii) Care should be used in applying this relieving method, because as the hover altitude changes, the maximum gross weight and power required (and the associated MGT) will vary. The data must be corrected to at least the maximum MGT for a minimum specification engine that can occur in service at the flight condition under investigation.

AC 29.1043A. § 29.1043 (Amendment 29-26) COOLING TESTS.

a. Explanation. Amendment 29-26 adds a new paragraph to § 29.1043(a)(5), to define "stabilization" as it pertains to powerplant systems cooling tests.

b. Procedures. All of the policy material pertaining to this section remains in effect with additional information that "stabilized temperatures" are achieved when the rate of change is less than 2° F per minute.

AC 29.1045. <u>§ 29.1045 CLIMB COOLING TEST PROCEDURES</u>.

a. <u>Objective</u>. The objective of the regulation is to verify, for Category A and for Category B rotorcraft described, that cooling provisions are adequate for a one-engine-inoperative (OEI) climb or descent initiated from a multiengine cruise at the critical altitude with stabilized component temperatures. The specific flight conditions and powers are described in the regulation.

b. <u>Explanation</u>.

(1) This regulation specifies climb or descent cooling with OEI for Category A rotorcraft and for Category B rotorcraft with Category A powerplant isolation and fireproof or isolated structure, controls, etc., which are essential for controlled flight and landing. For the Category B machine described, the testing should be accomplished at the steady rate of climb or descent established under § 29.67(b), i.e., at the best OEI rate of climb (or descent) and the remaining engine at maximum continuous power or 30-minute power, whichever is applicable.

(2) The engine whose shutdown has the most adverse effect on the cooling conditions for the remaining engine(s) and powerplant components should be inoperative.

(3) The regulation provides that the climb cooling test may be conducted in conjunction with the takeoff cooling test of § 29.1047. This possible combining of tests applies only to § 29.1047(a), since § 29.1047(b) is a multiengine climb and not related to the OEI climb procedures of § 29.1045.

c. <u>Procedures</u>.

(1) The OEI climb cooling test point begins from a multiengine cruise, with stabilized fluid and component temperatures, 1,000 feet below either the all-engine-critical altitude or the maximum altitude at which the rate of climb is 150 FPM, whichever is the lowest altitude. If the minimum altitude derived is less than sea level, the climb should begin from a twin engine cruise with stabilized fluid and component temperatures at the minimum practical altitude.

(i) The all-engine-critical altitude is the maximum altitude at which, for the ambient conditions prescribed, it is possible to maintain the multiengine specified power. For example, if for multiengine operations, the transmission maximum continuous torque can be maintained on the hot day to a maximum altitude of 10,000 feet above which power would have to be reduced because of gas temperature or other limitations, then 10,000 feet is the all-engine-critical altitude. Point ”A” in figure AC 29.1045-1 illustrates the all-engine-critical altitude.

(ii) The 150 FPM climb criteria should be based on multiengine operation at maximum continuous power available at hot day conditions at maximum gross weight.

(iii) Fluid and component temperatures are considered stabilized when the rate of change is less than 2° F per minute.

(2) The OEI climb power to be utilized is 30-minute OEI hot day power available (if approval of 30-minute power on the aircraft is requested), followed by maximum continuous hot day power available. If 30-minute OEI power approval is not requested, the power to be utilized would be maximum continuous hot day power available.

(i) Rotorcraft for which approval of a continuous OEI power rating is requested would use the power available on a hot day at the maximum continuous OEI rating following the 30-minute OEI climb phase (or for the entire climb if approval of 30-minute OEI power is not requested).

(ii) If the maximum continuous OEI approval is not requested, then the highest hot day power available approved for continuous usage from the remaining engine(s) under OEI conditions would be used following the 30-minute OEI climb phase (or for the entire climb if approval of 30-minute OEI power is not requested).

(3) In order to achieve representative test results, the rotorcraft climb rate and airspeed should approximate those which would occur on a hot day. This is accomplished by adjusting rotorcraft gross weight as required to produce the desired climb rate based on published or predicted climb performance data. The possible adverse effects of climb fuselage attitude on cooling air duct entrances should be considered in the selection of center-of-gravity of the test aircraft.

(4) The OEI climb should be continued for at least 5 minutes after the occurrence of the highest temperature recorded or until the maximum certification altitude is reached. Generally, temperatures would be expected to peak a short time after the climb begins since component and fluid temperatures are stabilized prior to entry to the climb phase.

(5) For Category B rotorcraft, defined in § 29.1045(a)(2) without a positive OEI rate of climb, the descent should begin from a hot day maximum continuous power multiengine cruise, with stabilized fluid and component temperatures, at the all-engine-critical altitude.

(6) The descent should conclude at either the maximum altitude at which level flight can be maintained with one engine inoperative or at the minimum practical altitude, whichever is higher.

(7) The OEI powers available to be utilized during the descent would be the same as those prescribed previously for OEI climb cooling. OEI operation should continue until component and fluid temperatures stabilize.

(8) The airspeeds utilized in the climb and descents should be representative of normal speeds unless cooling provisions are sensitive to rotorcraft airspeed, in which case the airspeeds most critical for cooling should be used. In no case, however, should it be required that the selected airspeeds exceed the speeds established under §§ 29.67(a)(2) and 29.67(b).

MULTIENGINE POWER AVAILABLE

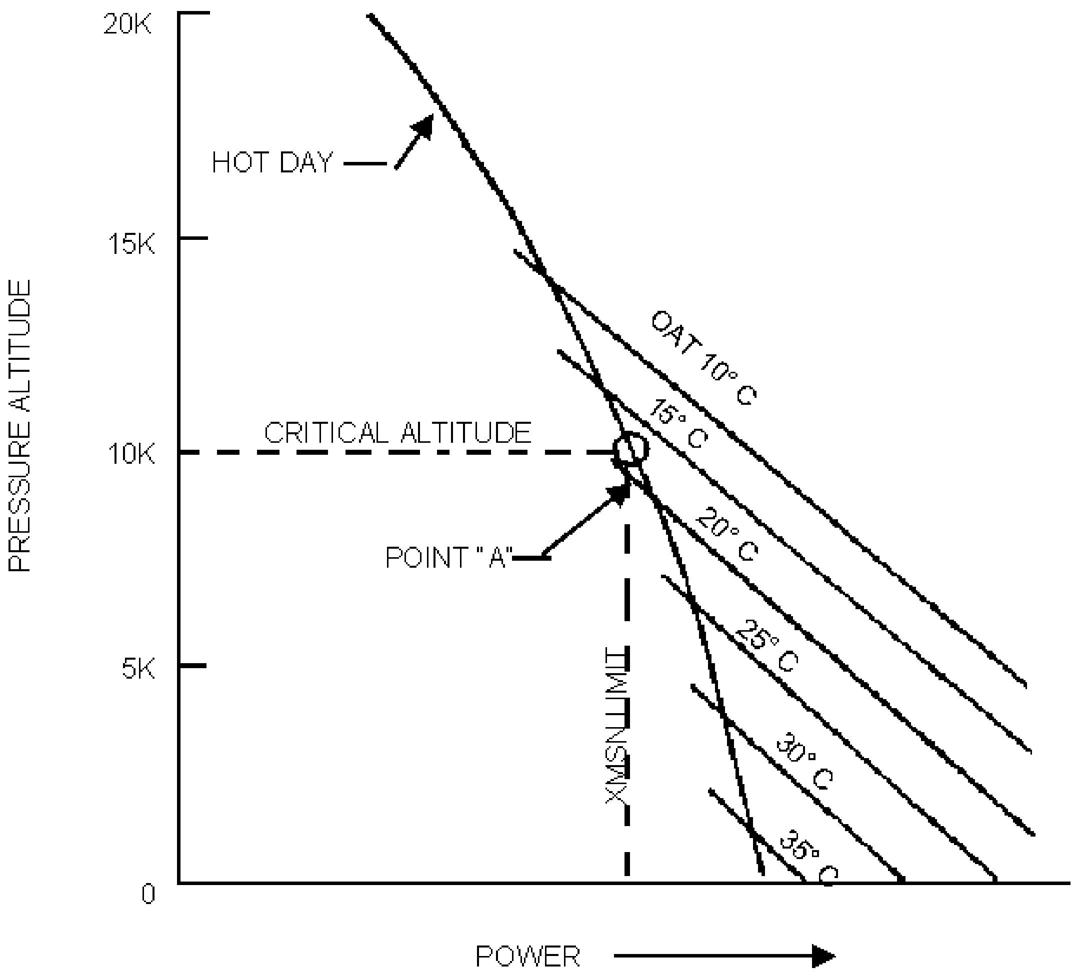


FIGURE AC 29.1045-1 ALL-ENGINE CRITICAL ALTITUDE ADDITIONAL COOLING TEST POINT

AC 29.1047. <u>§ 29.1047 (Amendment 29-1) TAKEOFF COOLING TEST PROCEDURES</u>.

a. <u>Objective</u>.

(1) For Category A rotorcraft, the objective is to verify satisfactory takeoff and OEI climb cooling for the Category A takeoff profile defined in aircraft performance §§ 29.59(c) and 29.67(a) following a prolonged hover.

(2) For Category B rotorcraft, the objective is to verify satisfactory cooling for the takeoff and subsequent climb for the Category B takeoff defined in performance §§ 29.63 and 29.65(a) following a prolonged hover.

b. <u>Procedure - Category A</u>.

(1) The rotorcraft is hovered in-ground-effect (IGE) at the power required to hover on the test day at the maximum Category A takeoff gross weight for the hot day, until temperatures stabilize.

(i) <u>Alternate Procedure</u>. If the test day OAT is high, it may not be possible to hover IGE at the prescribed gross weight without entering the takeoff range on the measured gas temperature (MGT) indicator. Since operations in the takeoff range are allowed only for 5 minutes and the typical stabilization time is 20 to 35 minutes, it is permissible to reduce the initial aircraft gross weight so the initial MGT will be at least at the MCP limit, but will not be in the takeoff range for more than 5 minutes; and

(ii) The fuel burn during the anticipated 20 to 35 minute stabilization period may cause the aircraft to leave the prescribed hover IGE condition unless power is reduced or additional weight is added by fluid transfer or other methods. It is permissible to reduce power to maintain the IGE hover for this phase of testing rather than attempt special weight control procedures.

(2) After temperatures have stabilized, an OEI climb is initiated from the lowest practicable altitude.

(i) Multiengine power may be used from the stabilized IGE hover to the CDP before OEI operations for cooling verification begin.

(ii) Actual shutdown of the simulated failed engine may not be necessary if the applicant can show that cooling of the remaining engine, fluids, and components is not affected by operation of the “failed” engine at idle power.

(iii) The power utilized at the initiation of the OEI climb would be the same as for establishing the takeoff climbout path of § 29.59, typically 2.5-minute OEI hot-day power available.

(3) After the time period for which the power is used in establishing the takeoff climbout path has expired, OEI power is changed to that used in meeting the steady rate of climb (150 FPM, 1,000 feet above the takeoff surface of § 29.67(a)(2)).

(i) The power to be used for this phase is 30-minute OEI hot-day power available, if approval of this power rating for performance is requested.

(ii) If 30-minute OEI approval is not requested, the highest hot-day power available approved for continuous usage under OEI conditions would be utilized.

(4) Climb at the OEI power used in meeting § 29.67(a)(2) would continue for at least--

(i) Thirty minutes if 30–minute OEI power is used; or

(ii) Five minutes after the occurrence of the highest temperature recorded, if other than 30-minute OEI is used.

(5) Unlike § 29.1045, the procedure set forth in § 29.1047 for Category A rotorcraft does not specifically require continuation of the OEI climb beyond the 30-minute duration allotted for 30-minute OEI power usage.

c. Procedure - Category B.

(1) The rotorcraft is hovered IGE until temperatures stabilize at the power required on the test day to hover IGE at the maximum Category B takeoff gross weight for the hot day.

(i) Alternate Procedure. If the test day OAT is high, it may not be possible to hover IGE at the prescribed gross weight without entering the takeoff range on the MGT indicator. Since operation in the takeoff range is allowed only for 5 minutes and the typical stabilization time is 20 to 35 minutes, it is permissible to reduce the initial aircraft gross weight so the initial MGT will be at least at the MCP limit, but will not be in the takeoff range for more than 5 minutes; and

(ii) The fuel burn during the anticipated 20 to 35 minute stabilization period may cause the aircraft to leave the prescribed hover IGE condition unless power is reduced or additional weight is added by fluid transfer or other methods. It is permissible to reduce power to maintain the IGE hover for this phase of testing rather than attempt special weight control procedures.

(2) After temperatures have stabilized in hover IGE, a multiengine climb is initiated at hot-day takeoff power available from the lowest practicable altitude. Section 29.1047(b)(3) requires only that takeoff power be maintained for the same time interval as used in determining the takeoff flight path under § 29.63. This time interval

could be less than the 5 minutes for which takeoff power is approved. Unless the applicant can show that the time interval used in § 29.63 provides more conservative results, or unless additional testing is proposed, the full 5 minutes allowed for takeoff power should be used to assure that the most critical condition has been surveyed.

(3) After the use of takeoff power for the appropriate time interval, the power should be reduced to multiengine maximum continuous hot-day power available and the climb continued until at least 5 minutes after the occurrence of the highest temperature recorded.

(4) The airspeeds utilized in the climb should be representative of normal speeds unless cooling provisions are sensitive to rotorcraft airspeed, in which case the airspeed most critical for cooling should be used. The airspeed need not exceed the speed for best rate of climb with maximum continuous power available.

AC 29.1049. § 29.1049 HOVERING COOLING TEST PROCEDURES.

a. Objective. The objective is to verify satisfactory hover IGE cooling at sea level and at the hover ceiling for hot-day conditions.

b. Explanation. The rule provides for a hover IGE cooling check in still air at sea level and at the hover ceiling at maximum continuous power. Still air is interpreted as a wind speed of 5 knots or less.

c. Procedures.

(1) The aircraft should be hovered IGE at the maximum certificated hover weight or at the IGE hover weight corresponding to hot-day maximum continuous power available, whichever is less.

(i) The power utilized would normally be hot-day maximum continuous power available and the initial gross weight would be selected as required to achieve hover IGE on the test day.

(ii) After initiation of the hover, special weight control procedures need not be implemented in attempting to maintain hover IGE as fuel burn-off occurs. The power may be gradually reduced to maintain the IGE hover condition.

(2) The hover test is to continue until at least 5 minutes after the occurrence of the highest temperature recorded.

(3) Section 29.1049 also requires a hover IGE at the maximum continuous power available at the altitude resulting in zero rate of climb.

(i) Often, compliance is illustrated by extrapolating component cooling margins from sea level test results and from selected altitude test site results to the altitude resulting in zero rate of climb.

(ii) Considerable engineering judgment must be exercised in utilizing the extrapolation method described. In general, if test data is extrapolated more than 2,000 feet to the hover ceiling from the highest altitude site selected and the resulting component margin is less than 5° F, additional verification at altitude may be required.

SUBPART E - POWERPLANT

INDUCTION SYSTEM

AC 29.1091. § 29.1091 (Amendment 29-17) AIR INDUCTION.

a. Explanation.

(1) The air induction system for each engine and auxiliary power unit must supply the air required under the operating conditions for which certification is requested. For reciprocating engine installations, the system must provide air that is suitable for proper fuel metering and mixture distribution. This should be shown with the induction system valves in any position.

(2) The intake system shall be designed such that a backfire flame will not constitute a fire hazard within the engine accessory compartment or within other areas of the powerplant compartment.

(3) Each reciprocating engine must have an alternate air source which must be located to prevent entrance of rain, ice, or other foreign matter.

(4) For rotorcraft powered by turbine engines and rotorcraft incorporating auxiliary power units, there must be means to prevent leakage of hazardous amounts of flammable fluids from entering the engine or auxiliary power unit intake system.

(5) Also, the air ducts must be located or protected to minimize the ingestion of foreign matter during takeoff, landing, and taxiing.

b. Procedures.

(1) For turbine-engine installation, the induction system should supply air of suitable quality to meet the installation requirements of the engine manufacturer. The installation requirements should be met throughout the operating envelope of the rotorcraft. In addition, the design and location of the air induction system should prevent accumulations of rain or hail, either external or internal to the induction system, that could adversely affect engine operation.

(2) The inlet design should account for the prevention of hazardous fluids entering the engine. Some designs will have inlet ducts which are free from any fluid lines; however, other designs may route the engine inlet air through a compartment which has flammable fluid lines. When this condition exists, test demonstrations of critical leakage during operation have been used to substantiate the installation. The fluid leakage may not have an adverse effect on engine operation.

(3) The air induction system design should also account for and minimize the possibility of foreign matter ingestion during takeoff, landing, and taxiing.

(4) For reciprocating engine installations, the induction system should supply air of suitable quality and quantity to the combustion system of the engine. The condition of this air at the entering face of the carburetor is extremely important. For proper operation, it is essential that the airflow be smooth and uniform, clean, and unrestricted throughout the very wide range of horsepower expected from the engine.

AC 29.1093. § 29.1093 (Amendment 29-22) INDUCTION SYSTEM ICING PROTECTION.

a. Reciprocating Engines. No advisory material is presented here for reciprocating engines since it is unlikely that these types will be used in transport rotorcraft.

b. Turbine Engines - Ice Protection.

(1) Explanation.

(i) This rule requires turbine engines and turbine-engine inlets to perform satisfactorily in atmospheric icing conditions defined in Appendix C of Part 25. On an equivalent safety basis, the limited icing envelopes described in paragraph AC 29.877 herein may be used to show compliance with the intent of the regulation if the rotorcraft is limited to not greater than a 10,000-foot pressure altitude for all operations. If operations are permitted above 10,000 feet, the Appendix C, Part 25, envelope must be used from 10,000 feet to the service ceiling or 22,000 feet. These possible equivalent safety approaches are not discussed herein. Compliance with the induction system icing protection rule is required regardless of flight manual limitations or restrictions against flight into atmospheric icing conditions.

(ii) In showing compliance with § 29.1093(b)(1)(i), the FAA/AUTHORITY has accepted the concept of limited exposure associated with escape from inadvertent ice encounters.

(A) It is presumed that there will be a flight manual limitation against flight into known icing, and that the engine induction system will be reevaluated if total aircraft ice protection certification is requested. Under this concept, the rotorcraft is assumed to fly directly through the icing environment; i.e., direct sequential penetration and straight line exit from both the continuous maximum and intermittent maximum icing clouds. Thus, the duration of exposure to the icing environment could be calculated by knowing the aircraft flight speed and cloud horizontal extent. A range of engine power and rotorcraft airspeeds should be evaluated to encompass the operating envelope of the rotorcraft.

(B) When this limited exposure concept is used, the aircraft type certificate data sheet should clearly specify that the engine induction system must be reevaluated if certification to the general ice protection regulation, § 29.877 or § 29.1419, is requested. This direct penetration and exit approach is inappropriate for aircraft for which full icing clearance is requested (reference § 29.1419).

(iii) Engine induction system continuous icing protection would be necessary for aircraft for which full-icing clearance is requested (reference § 29.1419(d)). The approach is much preferred for all programs in order to reduce the scope of any eventual total aircraft icing program effort and to increase the safety level in conducting the rotorcraft natural icing tests. Since at least one rotorcraft has been FAA/AUTHORITY certificated to operate in known icing conditions and others have active development programs to this end, applicants should anticipate eventual full-icing clearance and consider that the engine induction system may be required to operate routinely in a continuous icing environment.

(iv) It is noted in paragraph AC 29.877 that some natural icing tests are required to show compliance with the overall rotorcraft ice protection requirements. It is not required that the engine induction system be evaluated as a part of that natural icing test if adequate verification has been shown by tunnel testing, analysis, or other means to assure satisfactory operation in an extended continuous icing environment. If, however, subsequent rotorcraft natural icing testing shows unanticipated detrimental engine inlet effects, the inlet ice protection system should be reexamined.

(v) The regulation specifies the examination of flight idling conditions. This requirement is normally associated with a low-power letdown at the minimum practical forward airspeed. Alternatively, evaluation of the minimum power and minimum airspeed combination specified in the RFM for operation in visible moisture when below 40° F will accomplish the intent of the idling requirement.

(vi) An acceptable approach to a finding of compliance would be a combination of analysis of the performance of the ice protection system which covers the range of the applicable icing flight envelope (maximum altitude, minimum temperature, etc., of the basic rotorcraft) supported and validated by tests. Ideally, these tests would be conducted in natural atmospheric ice with special instrumentation for droplet size and liquid water content. In practice, however, natural icing testing may pose unacceptably severe problems since rotorcraft may not have the range and speed to reasonably find icing clouds and may not be equipped with the airframe and rotor ice protection needed for safety during the testing.

(vii) Problems with analysis emerge if engine inlets incorporate screens, turning vanes, sideward or upward openings, and edge or lip configurations which deviate from the airfoil shapes assumed in most of the analytical procedures described in current technical literature. The applicant should recognize that if meaningful analytical methods are not available, extensive testing with significant conservatism or possibly design changes may be required. Inlet screens in particular, if not adequately heated, fall in this category and can only be accepted if shown by very conservative ice testing to not significantly impede airflow to the engine.

(2) Procedures.

(i) Review paragraph AC 29.877, ADS-4, Report No. FAA-RD-77-767, and Advisory Circular 20-73. (The comparative concept described under Item 34 of AC 20-73 is obsolete and should not be considered.) These data provide extensive description and methodology for evaluation of ice protection systems, however, as noted above, these data generally apply to near straight line droplet trajectory with impingement onto conventional airfoil shaped inlets. As such, the applicability of these data to rotorcraft engine inlet ducts is limited and may require extensive adjustment to accommodate the different inflow trajectories and shapes of rotorcraft.

(ii) An analysis, appropriate to the configuration; i.e., heated or unheated impingement surfaces, should be prepared. To be acceptable, this analysis should show the inlet to be adequately protected by heat, or if unheated, to show that the inlet with ice accretions as predicted, will provide adequate airflow to the engine throughout the flight envelope of the rotorcraft.

(A) For heated surfaces, ADS-4 and Report No. FAA-RD-77-76 provide detailed suggestions on heat transfer analysis particularly applicable to bleed air heated inlet lips formed in airfoil shapes. These data are limited in applicability and may not be useful for analyzing engine inlet water droplet trajectories to be expected at low airspeed and high engine airflow. Actual icing tests may be needed to derive the impingement patterns for these conditions.

(1) Acceptability criteria for heated inlet ducts usually require sufficient heat to evaporate the water to be expected in a “continuous maximum” icing cloud and to anti-ice the duct during flight in “intermittent maximum” icing clouds, providing the run-back and refreeze to be expected does not cause additional airflow disruption or damage to the engine. Full-scale inlet icing tests with the engine installed and operating should be conducted to verify the analysis. Engine power changes which may be expected in service should be included in the testing. Wind tunnels equipped for icing tests probably are the most useful means of conducting these tests if natural icing tests are impractical. The rotor downwash effect should be considered to the extent possible by adjusting the inflow angle in the tunnel.

(2) The power loss (bleed air, generator load, etc.) attributable to the heating requirements will affect the performance of the rotorcraft. Normally, this may be accounted for by specifying a gross weight incremental deduction from the flight manual performance data for flight into visible moisture below 40° F.

(3) Special evaluation of the possibility of ice ingestion damage to the engine should be made for heated systems which considers the ice ingestion to be expected when the anti-ice system is actuated after a delay of 1 minute for the pilot to

recognize that the rotorcraft has encountered ice. This time delay may be reduced if the crew is provided adequate distinctive cues to alert them that the rotorcraft has encountered icing conditions.

(B) For unheated inlets, an acceptable method for showing compliance would include an extensive, detailed analysis (which shows that ice accretions on and in the inlet do not obstruct adequate airflow to the engine) and tests as necessary to validate the analysis. The analysis of ice accretion becomes even more questionable since the unheated inlet involves ice buildups which themselves progressively change shape during icing exposure.

(1) Flight testing with an instrumented rotorcraft in natural ice to verify the analysis is desirable; however, wind tunnel tests as discussed above may be used. Since unheated inlets typically continue to accrete ice as a function of exposure, both the analysis and the test should realistically consider the actual exposure to be expected in service. This should not be less than penetration of the continuous maximum icing cloud followed immediately by exposure to the intermittent maximum cloud for rotorcraft not certified for icing. Engine power changes which may be expected in service should be included in the testing, and a warm-up period at the conclusion of the icing exposure should be shown for some selected test points to evaluate potential ice breakaway and ingestion.

(2) For the nonicing certified rotorcraft using the limited icing exposure concept for inlet certification, some conservatism should be applied to account for the fact that inlet icing may occur without airframe icing, and that the escape procedure from this unapproved operating condition is not defined. A demonstration of 30-minute hold capability in the continuous maximum cloud would be acceptable. Alternatively, if positive cues (perhaps a carefully located ice detector) of potential inlet icing are provided to the crew, the time increment could be reduced to recognition plus 15 minutes (15-minute escape time after recognition is consistent with the single ice protection system failure recognition and escape guidance for aircraft ice protection systems in paragraph AC 29.877). It should not be assumed that airframe icing will always be available as a cue to potential inlet icing. The main rotor, for example, may not show icing indications above 25° F, whereas some inlets may ice critically near 32° F ambient. A reduction of the acceptable 30-minute exposure should not be based on observation of ice accretions on protruding components which are likely to be changed. For example, a limited exposure inlet icing program which reduces the inlet icing exposure time based on crew recognition of icing on the windshield wipers may be invalidated at a later date if a new windscreen deletes the wipers.

(iii) Inlet capability during IGE hover in icing conditions has not generally been considered for rotorcraft not certified for icing. Recently, however, the FAA/AUTHORITY is aware that some inlets may ice at zero airspeed near 32° F with no indications of airframe icing in the field of view of the crew. This special concern of operating within RFM limitations, and yet placing the induction system in jeopardy, may be addressed in several ways. If the induction system ice protection scheme is not dependent on airspeed for proper function, the issue may be addressed by tunnel testing with inlet airflows approximating hover with no particular attention to tunnel windspeed. For protection schemes which may be sensitive to airspeed (external screens have shown this tendency), actual hover demonstration at or near zero speed tunnel conditions may be appropriate. Icing detectors located to indicate induction system icing in hover may be an option to a hover icing protection demonstration. Recently, on an external screened configuration, the FAA/AUTHORITY has accepted a satisfactory IGE hover demonstration of 30 minutes at the critical ambient temperature (i.e., ambient consistent with no airframe icing but potential inlet icing), 0.6 grams/meter^3 LWC and 40 micron droplet size as an adequate response to this concern.

(iv) For aircraft requesting full icing approval, or for those electing to show continuous induction system icing protection, the forward flight icing exposure would not be less than that time required to stabilize any ice accretions observed during repeated cycles of the continuous maximum followed by intermittent maximum cloud exposure. Typically, any ice accretions resulting from these repeated cycles would be expected to stabilize in less than 30 minutes. The 30-minute hold capability in the continuous maximum icing environment could thus be assured without special testing by careful selection of the test points for this repeated cycle.

(v) A rotorcraft requesting full icing approval should also have hover capability in the icing environment. Intermittent maximum icing conditions are not likely to exist near ground level and a satisfactory demonstration could involve the ability to hover indefinitely in the continuous maximum icing environment. Alternatively, carefully worded RFM limitations to restrict hover time may be acceptable if the system is not capable of indefinite exposure. Hover capability verification may not involve zero airspeed demonstration if the inlet protection system is insensitive to rotorcraft airspeed.

(vi) The engine(s) must be installed or protected to avoid engine damage from ice ingestion due to ice accretion in the inlet or on other parts of the rotorcraft, including the rotors, which may break away to enter the inlet. If screens or bypass arrangements are provided for these purposes, they should be included in the icing tests and shown by test or rational analysis to effectively protect the engine.

(vii) For unheated inlets, significant ice accumulations to be expected on the inlet may adversely affect the engine stall margin, acceleration characteristics, duct loss, etc. Dry air flight tests to evaluate these aspects can be accomplished by affixing ice shapes to the inlet. These shapes should closely match the actual ice shapes defined by test or analysis.

c. Turbine Engines - Snow Protection.

(1) Explanation.

(i) Section 29.1093(b)(1)(ii) provides that the turbine engine and its air inlet system operate satisfactorily within the limitations established for the rotorcraft, in both falling and blowing snow. The section does not provide the definition of falling and blowing snow.

(ii) Since the regulation provides for certification “within the limitations established for the rotorcraft,” the FAA/AUTHORITY can accept a restriction against snow operations in the limitations section of the RFM in lieu of demonstration of compliance. If no restriction on snow operations appears in the RFM, it is presumed that the aircraft may operate in snow at the pilot’s discretion.

(2) Guidance.

(i) The FAA/AUTHORITY has accepted that engine induction system operation in falling and blowing snow can be approved without restriction if normal operations under the following conditions are demonstrated:

Visibility: ¼ mile or less as limited by snow.

Temperature: 25° F to 34° F (28° F to 34° F desired), unless other temperatures are deemed critical.

Operations: Ground operations - 20 minutes
IGE hover - 5 minutes
Level flight - 1 hour
Descent and landing

(ii) Rotorcraft Flight Manual visibility restrictions for falling and blowing snow operations are not appropriate.

(iii) Time limitations, other than possibly for ground and hover operations, are not appropriate.

(iv) Artificially produced snow should not be used as the sole means of showing compliance.

(3) Guidance Rationale.

(i) The test conditions specified--visibility, temperature, and operations--are based on previous certification programs, previous FAA/AUTHORITY guidance, and on research by the FAA technical center and others.

(A) Visibility. The test visibility defined, ¼-mile visibility or less as limited by snow, represents a heavy snowstorm and is the maximum likely to be encountered in service. Rotorcraft which have been certified to the ¼-mile visibility test criteria have not shown engine inlet snow-related service difficulties. It is important to note that the visibility specified is a test parameter rather than an operational limitation to be imposed on the rotorcraft after the tests are completed.

(B) Temperature.

(1) The ambient temperature specified is conducive to wet snow conditions. Wet snow tends to accumulate on unheated surfaces subject to impingement.

(2) Colder ambients, more conducive to dry snow conditions, may be critical for some induction systems. Colder exterior surfaces may be bypassed, and the snow crystals may stick to partially heated interior surfaces where partial melting and refreezing may occur.

(3) Company development testing or experience with very similar type induction systems may be adequate to determine the critical ambient conditions for certification testing.

(C) Operations.

(1) Ground running, taxiing, and IGE hover operations are generally the most critical since the rotorcraft may be operating in recirculating snow. Twenty-five minutes under these extreme conditions would seem a reasonable maximum, both from the view of pilot stress and the maximum expected taxi time prior to takeoff in bad weather.

(2) One hour of level flight operation under ¼-mile visibility snow conditions should provide ample opportunity for hazardous accumulations to begin to build.

(3) The descent and landing will provide an engine power change, an induction system airflow change, and a variation in the external airflow pattern near the induction system entrance. The initiation of the descent and final flare for landing may also produce additional airframe vibration transmitted to the induction system. These power, airflow, and vibration changes may provide an opportunity for any level flight accumulations to be ingested into the engine. Hazardous accumulations are not acceptable during or after any test phase.

(ii) Visibility may fluctuate rapidly in snowstorms. It is affected by the presence of fog or ice crystals, is not crew measured or controlled, and is difficult to estimate. A visibility operational limitation based on snow, therefore, is not appropriate.

(iii) Since during cruise in snow conditions the aircraft is likely to be in and out of heavy snowfall, it is not practical for the crew to account for the time spent in snow in level flight conditions. Thus, it is not appropriate to include time limitations in the RFM for level flight snow operations.

(iv) A practical ground and IGE hover time limitation of less than 25 minutes in recirculating snow may be considered. The expected action at the expiration of this specified time period would be shut down and inspection of the inlet system or transition to a safe flight condition where demonstration has shown that moisture accumulations will not intensify or shed and cause engine operational problems.

(v) Artificially produced snow is an excellent development tool and has been successfully used to indicate potential problem areas in induction systems. These devices are usually restricted to use for hover and ground evaluations, and the snow pellets produced by these machines are not sufficiently similar to natural snowflakes to justify the use of artificial snow as the sole basis of certification.

(4) Procedures.

(i) Satisfactory demonstration of the test conditions requires that the engine, induction system, and proximate cowling surfaces remain free of excessive snow, ice, or water accumulation. Excessive accumulation is defined as accumulation that may cause engine instability, damage, or significant loss of engine power. If a questionable amount of snow or moisture accumulates in the inlet, the applicant may elect to demonstrate that this amount in the form of snow or water and ice, as appropriate, can be ingested by the engine without incurring surge, flameout, or damage.

(ii) The conditions specified assume actual flight demonstration in natural snow. The ground operations and IGE hover test conditions assume operation in recirculating snow. Blowing snow, resulting from rotor airflow recirculation, can be expected to be more severe than natural blowing snow if the rotorcraft continues to move slowly over freshly fallen snow. Thus, the blowing snow operational capability is usually demonstrated by the taxi and hover operations in recirculating snow.

(iii) For VFR rotorcraft, the airspeeds for the level flight test condition should include the maximum consistent with the visibility conditions. For IFR operations, the airspeed should be the maximum cruise speed or the maximum speed specified for snow operations in the flight manual limitations, unless other airspeeds are deemed more critical. It is recognized that many rotorcraft initially certified VFR are later IFR certified with a resulting possible increase in airspeed in snow conditions. This factor should be considered if IFR certification is anticipated.

(iv) The visibility specified assumes that visual measurements are made in falling snow in the absence of fog or recirculating snow by an observer at the test site outside the tests rotorcraft's area of influence. An accepted equation for relating this measured visibility to snow concentration is $V = 374.9/C^{0.7734}$ where C is the snow concentration (grams/meter^3) and V is the visibility (meters).

(A) This equation can be reasonably applied to all snowflake type classifications and is credited to J.R. Stallabrass, National Research Council of Canada.

(B) Other equations may be applied if they are shown to be accurate for the particular snowflake types for the test program.

(v) The snow concentration corresponding to the visibility prescribed, ¼ mile or less, will be extremely difficult to locate in nature. Data from Ottawa, Canada, research indicate that fewer than 4 percent of the snowstorms encountered there meet the 0.91 grams/m^3 concentration associated with the ¼-mile visibility. Furthermore, the likelihood that the desired concentration will exist for the duration of the testing is even more remote. Because of these testing realities, it is very likely that exact target test conditions will not be achieved. Those involved in certification must exercise good judgment in accepting alternate approaches.

(vi) For some engine induction systems, it may become apparent by inspecting for moisture accumulations that ground and IGE hover operations in recirculating snow are much more severe than the level flight test. In this instance, it is reasonable to accept prolonged IGE operations in recirculating snow and to accept durations of less than 1-hour level flight in ¼-mile or less visibility. Best efforts should be made to assure that at least some level flight time is accomplished at ¼-mile or less visibility to assure that the spectrum is covered.

(vii) It should be determined that the visibility established at the test sight is limited by snow and not by fog or poor lighting (twilight) conditions.

(viii) The concentration of snow approaching the inlet in severe recirculation will far exceed the quantity encountered in the natural snowfall. Recirculation is necessarily a qualitative judgment by the test pilot. The snow concentration at the inlets during recirculation would vary for different rotorcraft types and would be dependent on rotor characteristics, power setting, and inlet location. For test purposes, recirculation should be the highest snow concentration attainable in the maneuver, or that corresponding to the lowest visibility at which (in the pilot's judgment) control of the rotorcraft is possible in the IGE condition. The visibility specification of ¼ mile or less outside of the recirculation influence becomes inconsequential provided that fresh, loose snow is continually experienced during the ground operation and IGE hover testing phase. However, since it is intended that the test phases be accomplished sequentially to assure that transition to takeoff and other transients are considered, the conditions at takeoff, level flight, and descent and landing should approximate the ¼-mile visibility criteria.

d. Turbine Engines - Ground Icing.

(1) Explanation. This requirement addresses the situation where extended ground operation in icing exposes the rotorcraft and its engine inlet to icing (ground fog)

conditions which may have different droplet impingement patterns and involve different and/or less effective means of ice protection. Note that the requirement is effective at Amendment 10 and is applicable regardless of any desire to prohibit dispatch into known icing conditions.

(2) Procedure. Since this condition assumes zero airspeed, wind tunnel testing may be inappropriate unless conservative extrapolation of low speed tunnel data can be determined to be valid. For protection schemes which are dependent primarily on airspeed for proper functions (external screens have shown this tendency), it may be necessary to verify adequate ground operation protection capability by very low speed tunnels or by the use of outside facilities such as the Canadian National Research Council's spray rig at Ottawa, Canada. For heated systems or for internal bypass schemes, tunnel speed may not be important, and adequate demonstration may be accomplished at higher tunnel speeds provided that internal inlet airflows and heat available are properly considered. Testing should proximate the regulatory test conditions and be continued for 30 minutes using engine power and control manipulation as normally accepted during taxiway operations, followed by an acceleration to takeoff power. The test time may be shortened if de-ice/anti-ice protection is adequate or if stabilization of ice build-up is affirmed. The induction system should be in condition for safe flight at the conclusion of the test.

AC 29.1093A. § 29.1093 (Amendment 29-26) INDUCTION SYSTEM ICING PROTECTION.

a. Explanation. Amendment 29-26 clarifies that the phrase, "within the limitations established for the rotorcraft" applies only to the requirement in § 29.1093(b)(1)(ii) for demonstrating flight in falling and blowing snow.

b. Procedures. All of the policy material for this section remains in effect with the update that turbine engines and turbine engine inlets should perform satisfactorily in atmospheric icing conditions defined in Appendix C of FAR 29 instead of FAR 25. In addition to paragraph AC 29.1093, the following procedures should be followed:

(1) A "serious loss of power" in this section has been interpreted to be any power loss that requires immediate pilot action. In addition, the term "adverse effect on engine operation" in § 29.1093(b)(1)(ii) has been interpreted to be an effect that would prevent the engine from achieving rated aircraft flight manual performance (takeoff/climb/etc.). This term also includes effects on the engine induction system characteristics to an acceptable level established by the engine manufacturer (inlet distortion, etc.).

(2) It should be shown that rotorcraft that are prohibited from flight into falling and blowing snow can exit inadvertent entrance into those conditions without adverse effect upon the operating characteristics of the engine or the rotorcraft.

(3) For full flight capability into snow, both falling and blowing, it should be shown that each engine, and its inlet system, will operate satisfactorily throughout the flight power range of the engine and the operating limitations of the rotorcraft. It should be shown that any build-up or accumulation of snow will not reduce or block the flow of inlet air to the engine. Any accumulations that become dislodged should not affect engine operation.

AC 29.1101. § 29.1101 CARBURETOR AIR PREHEATER DESIGN.

a. Explanation. Each carburetor air preheater must be designed and constructed to:

(1) Ensure ventilation of the preheater when the engine is operated in cold air.

(2) Allow inspection of the exhaust manifold that it surrounds.

(3) Allow inspection of critical parts of the preheater itself.

b. Procedures. Although carburetors of some design and fuel injections are free from icing difficulties, the most common remedy is to preheat the air supply entering the carburetor. In this way, sufficient heat is added to replace the heat lost due to vaporization of fuel, and the mixing chamber temperature cannot drop to the freezing point of water. The air preheater is essentially a tube or jacket through which the exhaust of one or more cylinders is passed with the air flowing over the heated surface raised to the required temperature before entering the carburetor. A control for adjusting the preheater valve is installed in the cockpit so that heat may be applied only when actually required to prevent ice formation.

AC 29.1103. § 29.1103 (Amendment 29-17) INDUCTION SYSTEM DUCTS AND AIR DUCT SYSTEMS.

a. § 29.1103(a):

(1) Explanation. This paragraph is intended to require the design of induction system ducts for engines and auxiliary power units to include fuel and water drains which are effective in the ground attitude and do not discharge into any location where the fuel drainage could be ignited to cause a fire hazard.

(2) Procedures. Determine that each induction duct is provided with at least one drain of sufficient size to minimize clogging and located at the low point of the duct with the rotorcraft in the ground attitude. Discharge from the drain should not create a hazard to the rotorcraft.

b. § 29.1130(b):

(1) Explanation. This paragraph applies to reciprocating engines and is intended to require the induction system to withstand the stresses of explosive backfire which must be expected in these engines.

(2) Procedures. The magnitude of the backfire to be considered is somewhat subjective; however, the rule can generally be satisfied by testing which involves inducing actual backfires in the engine. This can usually be accomplished by crossing ignition leads between cylinders to cause ignition when the intake valve is open. Tests should include both engine cranking and power-on regimes.

c. § 29.1103(c):

(1) Explanation. Induction ducts, particularly on reciprocating engines, involve connections with other ducts and with structure. Flexibility is required to prevent relative motion (expansion, structural deflections, etc.) from prestressing the duct.

(2) Procedures. Review the design for long runs of ducting between the engine and structural supports and between other connections or supports in the duct system. Short segments of the duct constructed of bellows will usually provide the necessary flexibility.

d. § 29.1103(d):

(1) Explanation. The effectiveness of fire extinguisher systems is based, in part, on testing for agent concentration in the fire zone with the airflows to be expected. Any duct failure (burnout) during an engine compartment fire may be expected to introduce air to dilute the agent concentration, or if the duct passes through a firewall, duct burnout could result in an opening in the firewall. Fireproof ducts, as specified by this rule, are needed to ensure the integrity of the firewalls and the effectiveness of the fire extinguisher system. Fire resistant ducts may be used if located totally within the fire zone.

(2) Procedures. Ducts within a fire zone are usually engine air induction ducts, air bypass ducts, or cooling air ducts. For ducts which penetrate the firewall or other fireproof construction such as fireproof cowling, verify that the duct is of fireproof construction. Other ducts may be only fire resistant. A duct constructed of material which has been accepted as firewall material would be considered as fireproof without further testing (unless the duct is subject to significant structural loads, in which case, fire testing may be necessary with the loads applied to the duct). The tests for “fireproof” and “fire resistant” qualification differ only in the time exposure; i.e., 15 minutes for “fireproof” and 5 minutes for “fire resistant.” If nonmetallics are used in duct construction intended for “fireproof” applications and the integrity of the test specimen is deteriorating towards the end of the 15-minute fire test period, assessment of the situation with respect to possible hazards if the engine fire were to exist beyond 15 minutes is appropriate. Duct burnout should not result in the possibility that fire could escape the fire zone and create hazardous conditions.

e. § 29.1103(e):

(1) Explanation. This rule requires additional fireproofing of the inlet duct of auxiliary power units (APU's) to ensure safe disposal or containment of hot gas reverse flow from the APU from entering any other compartment of the rotorcraft in which a hazard would be created. This rule could, in some designs, require fireproof construction of the inlet duct for the APU to extend upstream beyond the confines of the firewall provided in compliance with § 29.1191(b). The extent of the fireproofing is subjective and may require malfunction testing if no applicable information can be provided by the manufacturer of the APU. For ducting upstream of the fireproof section, materials selected need not be qualified for fire impingement; however, they must be shown to be suitable for the maximum normal heat conditions to be expected.

(2) Procedures. Normally, fireproof ducting upstream of the APU to the contour of the rotorcraft is acceptable for compliance. However, if this distance is less than 36 inches, the possibility of impingement of hot gases on the contour skin of the rotorcraft is required. Fireproofing of contour skin or duct relocation should be considered if the impingement area is a nonmetallic structure or is part of or close to fuel tanks. Other system air inlets in the impingement area should also be evaluated for possible hazards due to ingestion of hot gases in event of reverse flow from the APU.

f. § 29.1103(f):

(1) Explanation. APU inlet ducts subject to reverse flow of hot gases should be constructed of materials that will not absorb fuel or other flammable liquids to avoid induction duct inlet fires which may ignite by backfires or reverse flow from an APU.

(2) Procedures. Any nonmetallic duct material should be shown by test or by previous qualification to be sealed or otherwise free of tendencies to absorb flammable liquids. Tests, if necessary, should follow the guidelines for absorption qualification set forth in TSO's or military specifications for fuel and oil tanks.

AC 29.1105. § 29.1105 INDUCTION SYSTEM SCREENS.

a. Explanation. This paragraph concerns reciprocating engine installations. If induction system screens are used, the following considerations apply.

(1) Each screen must be upstream of the carburetor.

(2) No screen may be in any part of the induction system that is the only passage through which air can reach the engine unless it can be deiced by heated air.

(3) No screen may be deiced by alcohol alone, and it must be impossible for fuel to strike any screen.

b. Procedures. Inlet screens in the engine induction system are generally provided to prevent the entrance of foreign objects. The induction design may incorporate features which address the concerns identified above. Also, some designs incorporate an alternate air door which, with appropriate consideration, accounts for the requirements of this paragraph. The alternate air source should provide the required air to maintain flight and landing to a suitable landing site at appropriate airspeeds and gross weights.

AC 29.1107. § 29.1107 INTERCOOLERS AND AFTER-COOLERS.

a. Explanation. Each intercooler and after-cooler must be able to withstand the vibration, inertia, and air pressure loads to which it would be subjected in operation.

b. Procedures. In complying with this regulation, the various vibrations, inertia, and air pressure loads should be identified. The installation may be verified by either analysis or test appropriate for the design.

AC 29.1109. § 29.1109 CARBURETOR AIR COOLING.

a. Explanation. It must be shown under § 29.1043 that each installation using two-stage superchargers has means to maintain the air temperature at the carburetor inlet, at or below the maximum established value.

b. Procedures. When the powerplant installation design utilizes a supercharger installation, it should be shown by testing that the air temperature at the carburetor inlet does not exceed established values.

SUBPART E - POWERPLANT

EXHAUST SYSTEM

AC 29.1121. § 29.1121 (Amendment 29-13) EXHAUST SYSTEM - GENERAL.

a. Explanation.

(1) This section addresses the arrangement of exhaust components and the protection against hazardous conditions which exist with hot exhaust gases for powerplant and auxiliary power unit installations.

(2) The objective is to ensure safe disposal of exhaust gases without fire hazard or physical impairment to any occupant.

b. Procedures.

(1) During the certification process, carbon monoxide levels should be monitored in the personnel compartments to verify that the gas levels are well within the acceptable range. The conditions under which the measurements are taken should be representative of the normal operating limitations of the rotorcraft. This paragraph is not applicable to gas turbine-engine-powered rotorcraft.

(2) Exhaust system surfaces hot enough to ignite flammable fluids or vapors must meet the isolation or shielding requirements of this section in addition to the requirements of §§ 29.1183 and 29.1185. Good design practice suggests that the isolation and shielding features incorporated would continue to be effective under the emergency landing conditions specified in § 29.561.

(3) Compliance with the § 29.1121(c) fireproof requirements can be accomplished by demonstrating that the material or component will withstand a 2000° F ± 50° F flame for 15 minutes while still fulfilling its design purpose. This testing should accurately simulate, as near as practicable, the operating environment of the material or component in service. In addition to the fireproof requirements, the requirements of § 29.1191 must be met.

(4) Compliance with § 29.1121(d) can be accomplished by locating the vents and drains where fumes and fluids cannot interact with the hot exhaust gases. Drains should discharge positively and be a minimum of 0.25 inches in diameter. No drain may discharge where it will cause a fire hazard. This can be demonstrated by discharging a colored liquid through the drain system in flight and on the ground. The dye should not impinge on any ignition source.

(5) It should be demonstrated that exhaust gases are discharged in such a manner that they do not cause distortion or glare seriously affecting the pilot's visibility

at night. One method of compliance would be a night flight evaluation at critical azimuth and variable wind conditions to verify that no degradation exists.

(6) Hot spots that can occur on exhaust system components should be eliminated by providing deflectors and/or adequate ventilation. Exhaust shrouds can either be ventilated or insulated to keep the temperatures low enough so that ignition of flammable vapors or fluids cannot occur under normal operation or under the emergency landing conditions specified in § 29.561.

(7) Compliance with § 29.1121(h) can be accomplished by ensuring that the drain will not discharge where it might cause a fire hazard. This can be demonstrated by discharging a colored liquid through the drain system in flight and on the ground. The dye should not impinge on any ignition source.

AC 29.1123. § 29.1123 EXHAUST PIPING.

a. Explanation. This section contains the following requirements that must be met for proper certification of exhaust piping on engines, auxiliary propulsion units (APU), and other similar devices.

(1) § 29.1123(a) requires that the piping be heat and corrosion resistant so that it performs its intended function during its operational life (either the life of the rotorcraft or a specified limited life) without significant metal corrosion, metal erosion, or creation of hazardous hot spots. The piping system should be designed, have an installation design, or a combination that allows performance of its function without thermal expansion (thermal strain) induced structural failures, such as ruptures caused by operating temperature excursions and by overpressurization during its operational life.

(2) § 29.1123(b) requires that the piping must be supported to withstand the vibration and loading environment (including inertia loads) to which it will be subjected in service.

(3) § 29.1123(c) requires that piping that connects to components between which relative motion exists in service must have the necessary flexibility and structural integrity to withstand the relative motion without exceeding limit load (at the maximum operating temperature) of the piping, or creating unintended loads (or load paths) on the components to which the piping connects.

b. Procedures. Exhaust piping is typically certified by analysis and installation tests conducted during the basic certification process, including flight tests, as follows:

(1) For compliance with § 29.1123(a), because of its durability in the hot exhaust environment, exhaust piping is typically made from stainless steel or alloy steel of the appropriate structurally and thermally derived wall thickness. Hot aircraft exhaust gases are very corrosive; thus, proper material selection and corrosion protective design should be performed and validated during certification. Advisory Circular

(AC) 43-4, “Corrosion Control For Aircraft” contains a detailed discussion of exhaust gas corrosion problems. Analysis and/or verification tests of the exhaust system should be conducted. This work is necessary to ensure thermal and structural integrity; to ensure that thermal expansion does not cause a structural overload or failure; and, to ensure that exhaust piping does not contact (or come close to) ambient temperature materials (such as structure or system components). Hot exhaust piping in contact with (or close to) ambient temperature materials can either create a fire hazard or cause an unintended strength reduction. To ensure that thermal expansion analyses and tests are properly conducted, the maximum in-service temperature excursion should be properly defined. The maximum temperature excursion should be based on the maximum temperature of the piping and exhaust gases, as affected by the insulatory characteristics of the piping’s enclosure, and as affected by a worst case hot day. The worst case temperature environment used for analysis can be verified by a temperature survey. If run on cooler days, the survey can be adjusted for the worst case hot day environment using methods identical to those used for engine cooling tests (reference paragraph AC 29.1043, Cooling Tests). The piping should be designed to expand freely so that thermal expansion (thermal strain) induced loads on the piping and its restraint system are minimized. If thermal expansion induced loads (in conjunction with deflection induced loads and exhaust flow loads, discussed in b(4)) are significant relative to limit load of any item in the load path, then a fatigue check on the critical design point(s) should be performed. The fatigue check should establish a safe life or an approved limited life for the critical component(s) in the system. An accurate analytical fatigue check on exhaust piping may be difficult to perform because of erosion, corrosion, etc., in service; therefore, phased inspections should be considered to ensure the exhaust piping’s continued airworthiness.

(2) For compliance with § 29.1123(b), exhaust piping should be properly supported so that the maximum loads anticipated in-service are properly distributed and reacted, and, as previously discussed, so that thermal expansion induced loading is minimized. Typically the worst case static design load conditions are either the inertia loads from an emergency impact (reference § 29.561) or the combined loading from thermal expansion, in-flight deflections and internal exhaust gas flow (See paragraph b(4)). It should be noted that several combinations of these loads should be examined to determine the critical combination. The piping should be supported and restrained such that critical frequencies are avoided and the induced vibration environment’s effect is minimized. Flight test vibration surveys may be necessary, in some cases, to properly define or validate the critical modes and environment and their effect on the exhaust piping design. Operating modes such as ground idle, flight idle, 40 percent and 80 percent of maximum continuous power, maximum continuous power, OEI power settings and other power settings should be investigated to determine their vibratory effect on the exhaust gas piping system. The strength reduction of the piping materials at operating temperature (and at worst case temperature) should be properly considered in the design and structural substantiation. MIL-HDBK-5D contains material allowables versus temperature data for a wide variety of metallic engineering materials.

(3) For compliance with § 29.1123(c), the piping and its restraint system should be designed to minimize loading induced on the piping by the relative motion (in-service deflections) of the components to which the system attaches. Isolation of significant deflection induced loading (if required based on analysis and strain surveys) by use of flexible joints or other equivalent devices or designs should be considered. Any such in-line device used to reduce deflection loading should be fireproof and leak free when performing its intended function.

(4) For critical load case determination, the expansion-induced thermal loading should be added in with mechanical relative-motion induced loads and internal exhaust gas flow loads to provide total critical loads for both a proper static and a proper fatigue structural substantiation. The critical combined static load should be compared with the emergency impact loads of § 29.561(paragraph b(2)) to determine the critical design load case for static strength substantiation.

(5) It should be noted that the majority of the exhaust piping verification testing required for certification can be accomplished during the rotor drive system tie down testing of § 29.923.

AC 29.1125. <u>§ 29.1125 (Amendment 29-12) EXHAUST HEAT EXCHANGERS</u>.

a. <u>Explanation</u>. This section applies only to rotorcraft powered by reciprocating engine(s) or equipped with reciprocating auxiliary propulsion units (APU). This regulation states the certification requirements for exhaust heat exchangers (EHE's) which are summarized as follows:

(1) § 29.1125(a) requires that each EHE be constructed and installed to withstand vibration, inertia and other operational loads.

(2) § 29.1125(a)(1) requires that each EHE be able to operate continuously at the highest anticipation service temperature.

(3) § 29.1125(a)(1) requires that each EHE be corrosion resistant to exhaust gases and other corrosion sources.

(4) § 29.1125(a)(2) requires that each EHE have provisions for inspecting its critical parts and areas.

(5) § 29.1125(a)(3) requires that each EHE have cooling provisions where it is subjected to hot exhaust gases.

(6) § 29.1125(a)(4) requires that each EHE muff design eliminate stagnation areas or liquid traps that would contribute to ignition of leaked flammable fluids.

(7) § 29.1125(b) requires that each EHE used to heat ventilating air for occupants--

(i) Either have a secondary heat exchanger between the primary EHE and the ventilating air system; or

(ii) Have other equivalent means to prevent harmful contamination of ventilating air.

b. Procedures. EHE's and their installations are typically certified by analysis and installation tests conducted during the basic certification process, including flight tests or simulated flight tests, as follows:

(1) Because of their durability in the hot exhaust environment, EHE's are usually constructed from stainless steel or alloy steel of the appropriate structurally and thermally derived wall thickness. The EHE and its system should be designed to expand freely to minimize thermal expansion (thermal strain) induced loads on the EHE and its restraint system. If thermal expansion induced loads (in conjunction with deflection induced loads and exhaust flow loads) are significant relative to the limit load of the EHE or its attachments, a fatigue check on critical design point(s) should be performed. The fatigue check should establish a safe life or an approved limited life for the critical component(s) in the EHE system.

(2) EHE's should be properly supported so that the maximum loads anticipated in service are properly distributed and reacted and so that thermal-expansion-induced loading is minimized. Typically, the worst-case static design load conditions are either the emergency impact loads acting alone (reference § 29.561), or the critical combination of loads from thermal expansion, in-flight deflections and internal exhaust gas flow. Several combinations of these loads should be examined to determine the critical combination. The EHE should be supported and restrained so that critical frequencies are avoided and the induced vibration environment is minimized. Flight tests or bench tests, such as vibration surveys conducted during rotor system endurance testing, may be necessary in some cases, to properly define or validate the vibration environment and EHE's critical modes and their effect on EHE design. Operating modes such as ground idle, flight idle, 40 percent and 80 percent of maximum continuous power, maximum continuous power, OEI power settings, and other critical power settings should be investigated to determine their vibratory effect on the EHE system. The strength reduction of EHE materials at operating temperature and at critical temperatures should be properly considered in EHE design and structural substantiation (MIL-HDBK-5D contains material allowables versus temperature data for a wide variety of metallic engineering materials). The EHE and its restraint system should be designed to minimize loads induced by the relative motion (in-service deflections) of the components to which the EHE attaches. Isolation of significant-deflection-induced loading (as required, based on analysis and strain surveys) by use of flexible joints, other equivalent flexible devices, or designs should be considered. Any such in-line device used to reduce deflection loading should meet applicable certification requirements and be leak-free.

(3) Expansion analysis and verification tests of the EHE should be conducted to ensure its thermal (and structural) integrity and to ensure that thermal expansion does not cause the EHE to contact (or come close to) ambient temperature aircraft materials, structure or system components and either create a fire hazard or an unintended reduction in strength. To ensure that expansion analyses and tests are properly conducted, the maximum in-service temperature excursion should be properly defined. The maximum temperature excursion should be based on the maximum temperatures of the EHE and exhaust gases, as affected by the insulatory characteristics of the EHE's enclosure, and as affected by a worst-case hot day. The worst-case temperature environment used for analysis can be verified by a temperature survey which, when run on cooler days, can be adjusted to the worst-case hot day environment using methods identical to those used for engine cooling tests (reference paragraph AC 29.1043, Cooling Tests).

(4) Hot aircraft exhaust gases are very corrosive; thus, proper material selection and corrosion protection design should be performed and validated during certification. Advisory Circular (AC) 43-4, "Corrosion Control For Aircraft" contains a detailed discussion of exhaust gas corrosion problems. The in-service corrosive environment should be identified and characterized as thoroughly as possible by chemical analysis, tests and service experience. Once defined, appropriate design techniques and materials should be selected. Certification tests may be required to ensure proper substantiation. Phased inspections and inspectability should be considered (reference (4)).

(5) The EHE's design should be reviewed for inspectability to ensure that structural and thermal integrity is maintained over the intended life of the EHE. Also, if the design review is not conclusive relative to inspectability, a tear down inspection should be conducted.

(6) Each EHE design should be reviewed, analyzed, and tested to ensure that cooling provisions are adequate where EHE surfaces are subjected to hot exhaust gases. This is necessary to prevent hazardous hot spots or a burn through which may cause a fire and contaminate the occupied environment.

(7) Each EHE design should be reviewed, analyzed, and tested to ensure that stagnation areas and liquid traps do not exist. This can be done using bench flow tests. These stagnant areas and traps could become ignition sources if wetted with a leaking flammable fluid. A review of potential leaking flammable fluid hazards should be conducted and appropriate preventative measures such as drains and drip fences installed to ensure they are routed away from EHE's.

(8) Each EHE design which will be used to heat ventilating air for occupants should be reviewed to ensure that the EHE is a double walled system, (i.e., it would require failure of two EHE surfaces to allow toxic exhaust gases to intermix with cabin ventilating air). Each EHE wall should be designed with equal thermal and structural resistance since a single undetected inner wall failure would subject the outer wall to

the primary heat load. Also, inspectability provisions should be provided or means identified to ensure that inner wall failures can be detected in service. Any equivalent means which is applied for must clearly provide an equivalent level of safety to a double walled EHE.

SUBPART E - POWERPLANT

POWERPLANT CONTROLS AND ACCESSORIES

AC 29.1141. §. 29.1141 (Amendment 29-13) POWERPLANT CONTROLS: GENERAL.

a. Explanation.

(1) Section 29.1141(a) References §§ 29.777 and 29.1555. The detailed compliance procedures for powerplant control arrangement and markings are found in these sections.

(2) Section 29.1141(b) requires that controls be located and/or shielded such that normal movement of cockpit personnel will not cause inadvertent control movements.

(3) Section 29.1141(c) requires that each flexible control (push-pull cables) be properly approved.

(4) Section 29.1141(d) requires that each control maintain its set position without movement from an inadvertent source such as vibration or control system loads. This is required so that constant flightcrew attention is not necessary.

(5) Section 29.1141(e) requires that each control be able to withstand operating loads without excessive deflection. Excessive deflection is interpreted to be that deflection that would cause erratic movement, lack of crispness, or premature failure.

(6) Section 29.1141(f) specifies acceptable open/close positions for manual valves to prevent power failure due to improper control valve positioning. Power-assisted valves should have means to indicate to the flightcrew that the valve is either in the fully open or fully closed position or that the valve is moving between these two positions.

(7) The control system is subject to evaluation under § 29.901(c); i.e., for turbine installations, no single failure or malfunction, or probable combination thereof, of any powerplant control system should cause the failure of any powerplant function necessary for safety. One acceptable way to determine this is by use of a failure modes and effects analysis (FMEA).

b. Procedures.

(1) For compliance with § 29.1141(a), review the procedures for paragraph AC 29.1555. Evaluation by the flight test pilot during the official flight test program is appropriate.

(2) Compliance with § 29.1141(b) is normally evaluated during the flight test program and documented in the flight test report.

(3) Compliance with § 29.1141(c) may be accomplished by qualifying the control to MIL-C-7958, “Controls, Push-Pull, Flexible, and Rigid,” or other approved standards or by previous approval in a similar function, installation, or arrangement.

(4) Compliance with § 29.1141(d) may be shown during the flight test program by monitoring the means to prevent control creep. This device or arrangement should be effective without crew attention and should not impose undue control displacement loads or interfere with accurate settings.

(5) Compliance with § 29.1141(e) may be shown by an appropriate structural analysis and/or a witnessed static load test using the factors specified under § 29.397 unless a lower value can be shown to be applicable. Operation tests and design details described in §§ 29.683 and 29.685 should also be considered.

(6) Compliance with § 29.1141(f)(1) may be accomplished by installing manual valves which have positive stops in the fully open and closed positions. The fuel valves, however, may have an arrangement to facilitate the capability of switching to different fuel tanks if suitable indexing is provided. Compliance with § 29.1141(f)(2) may be accomplished by installing a device which displays to the flightcrew one indication with valve fully open and another with the valve fully closed. Alternatively, an indication could be given when the valve is moving from fully open to fully closed with the indication ceasing when the valve position corresponds to the selected switch position (open or closed). An example would be a light that is “off” when the valve is fully open or fully closed and illuminates while the valve is transitioning.

AC 29.1142. <u>§ 29.1142 (Amendment 29.17) AUXILIARY POWER UNIT CONTROLS</u>.

a. <u>Explanation</u>.

(1) This section addresses control requirements for any APU installed in a rotorcraft.

(2) The requirement for starting, stopping, and emergency shutdown of the APU from the flight deck is primarily to control APU operation in the event of improper operation or malfunction which could affect the safety of the aircraft.

b. <u>Procedure</u>.

(1) The requirements of this section apply to all APU installations in rotorcraft without regard to whether or not the APU is to be operated on the ground only, or operated in flight and on the ground.

(2) The APU installation must provide sufficient controls to the flight crew to enable them to control the operation of the APU under normal and emergency conditions.

(3) Compliance can be shown by both demonstration and a failure analysis.

AC 29.1143. § 29.1143 (Amendment 29-12) ENGINE CONTROLS.

a. Explanation. This section prescribes safety standards applicable to arrangement and operation of the engine controls.

(1) Section 29.1143(a) requires a separate throttle for each engine.

(2) Section 29.1143(b) requires a throttle arrangement for control of all engines be achieved by:

(i) Separate control of each engine.

(ii) Simultaneous control of all engines.

(3) Section 29.1143(c) requires that immediate actuation at the engine control should be provided by any given input at the cockpit throttle control.

(4) Section 29.1143(d) requires that each fluid injection system control (e.g., water-alcohol) other than the fuel system control must reside in the throttle controls. This does not preclude the injection system pump from having a control located separately from the throttle.

(5) Section 29.1143(e) requires that power or thrust controls (that have fuel shut-off features) provide a means to prevent inadvertent movement to the shut-off position. This means should--

(i) Provide a positive lock or stop at the idle position; and

(ii) Require a separate and distinct operation to place the control in the shut-off position.

b. Procedures.

(1) Certification data submitted by the applicant should be reviewed to ensure that all the design features stated in § 29.1143 exist.

(2) Proper engine control functioning (to verify the design features of § 29.1143) should be verified as part of the type inspection authorization (TIA) for the certification project.

(3) Compliance with § 29.1143(e)(1) has been shown successfully in the past by use of idle detents (mechanical or electrical/mechanical such as a solenoid).

(4) In the past, compliance with § 29.1143(e)(ii) has been achieved by use of a switch or button to displace the idle stop or by use of distinct offsets in throttle motion to allow movement from the idle stop to shutoff.

AC 29.1143A. <u>§ 29.1143 (Amendment 29-26) ENGINE CONTROLS</u>.

a. <u>Explanation</u>. Amendment 29–26 revises § 29.1143 by replacing the terms “throttle control” and “thrust control” with the more general term “power control.” The changes should preclude misconceptions regarding engine control arrangements when governor-controlled turboshaft engines are employed in rotorcraft.

b. <u>Procedures</u>. The means of compliance for this section is unchanged.

AC 29.1143B. <u>§ 29.1143 (Amendment 29-34) ENGINE CONTROLS</u>.

a. <u>Explanation</u>. Amendment 29-34 introduced the option of using 30-second/2-minute OEI power ratings to multiengine rotorcraft. This amendment revises § 29.1143 by adding the requirement for automatic control of 30-second OEI limits in the new § 29.1143(e). Automatic control of the 30-second OEI limits are required to prevent exceedances of the remaining power sections after the precautionary shutdown of one engine. The use of 30-second OEI power must be limited to emergency use only during flight conditions where one engine has failed or has been shutdown for precautionary reasons. During this critical stage of flight crew attention should not be focused on powerplant instruments to avoid limit exceedances.

b. <u>Procedures</u>. The automatic controls used to prevent 30-second OEI limit exceedances can be installed on the airframe or the engine. The applicant should demonstrate that 30-second OEI limits that can affect the continued safe operation of the drive system or engine such as gas generator speed, measured gas temperature, torque, etc., cannot be exceeded. It should also be shown that these devices do not restrict the ability to achieve the full 30-second OEI limits. The operation of these limit devices can be demonstrated on the aircraft or if possible by using bench tests.

AC 29.1145. <u>§ 29.1145 (Amendment 29-13) IGNITION SWITCHES</u>.

a. <u>Explanation</u>.

(1) This section addresses the arrangement and protection of ignition switches for reciprocating engines or for turbine engines which require continuous ignition.

(2) The objective is to provide a means to shut off all ignition quickly, if required, while at the same time providing protection against inadvertent ignition switch operation.

(3) Section 29.1145(b) does not specifically state that turbine engines not requiring continuous ignition are excluded from the rule, but no benefit is realized by the capability of shutting off all ignition to these engines.

b. Procedures.

(1) Section 29.1145(b) is self-explanatory in specifying that a means be available to shut off all ignition quickly by the grouping of switches or by a master ignition switch control. A "T" arrangement or split rocker switches are possible configurations. A master ignition control, if utilized, would need to be carefully evaluated if rotorcraft performance credit is given for engine isolation.

(2) Each group of ignition switches and the master ignition control should have a means to prevent inadvertent operation. "Guarded" switches are the usual means of showing compliance.

AC 29.1147. § 29.1147 MIXTURE CONTROLS.

a. Explanation. This section addresses the arrangement of fuel mixture controls, if installed. Major manual adjustment of the fuel mixture to optimize performance is not normally allowed due to the possibility of engine failure or detonation if significant misadjustment occurs. If "best-power" with respect to fuel mixture is desired, normal practice is to utilize engines with automatic mixture controls, in which case the lever in the cockpit reverts to merely an engine shutdown device. In any case, manual adjustment of the mixture, except for intentional shutdown, should not be prescribed without positive means of ascertaining that the resulting fuel-air mixture is within the range associated with safe engine operation. Some manual mixture adjustment may be acceptable for more efficient engine operation if suitable stops or automatic means are provided to prevent inadvertent engine shutdown with mixture movement or engine malfunction with flight condition changes.

(1) Section 29.1147(a) requires (if mixture controls exist) that controls be arranged to allow:

(i) Separate control of each engine.

(ii) Simultaneous control of all engines.

(2) Section 29.1147(b) requires that each intermediate position of the mixture controls corresponding to a normal operating setting be identifiable by both feel and sight.

b. Procedures.

(1) Certification data submitted by the applicant should be reviewed to ensure that the design features stated in § 29.1147 exist.

(2) Proper mixture control functioning (to verify the design features of § 29.1147) should be verified as part of the TIA for the certification project.

(3) Compliance is typically shown by use of a side-by-side arrangement of the controls, provided that the arrangement is compatible with other controls and considering that crew attention to the primary flight controls may be a full-time, "hands-on" operation.

AC 29.1151. § 29.1151 ROTOR BRAKE CONTROLS.

a. Explanation.

(1) Paragraph (a) of § 29.1151 is intended to require design features which, for all practicable purposes, prevent brake application in flight even under conditions of reasonably expected crew error or confusion.

(2) Paragraph (b) of § 29.1151 would require warning devices to alert the crew if the brake has not been completely released.

b. Background. Inadvertent or undetected application of the rotor brake is expected to result in excessive heat and fire in the rotor brake area. Rotor brake components are usually located integral with, or in close proximity to, rotor drive system components and, in many cases, close to critical hydraulic main rotor control system components. Fires in these areas would be extremely hazardous.

c. Methods of Compliance.

(1) For paragraph (a) literal compliance can be achieved by lock-out devices sensitive to the higher RPM. ranges of the main rotor or other flight parameters, hydraulic bypass or lockout devices controlled by flyweight governor systems, etc. The guard required by § 29.921 does not, in itself, provide compliance with this requirement. For some designs, if careful evaluation of the overall control, including location, guard mechanism, control manipulation requirements, accessibility, etc., provides an extremely high degree of assurance that inadvertent application will not occur, compliance may be assumed. Also, if brake application does occur, annunciation appears, and no immediate hazard to flight operation exists, compliance may be assumed.

(2) Warning devices supplied to comply with this rule should provide a signal at any time the rotor brake is engaged, including partial engagement. Typically,

micro-switches installed to close a circuit to a cockpit warning (red) light when the brake puck moves out of the retract position will provide compliance, provided the designer gives full consideration to the vibration, temperature, moisture, and other environmental considerations appropriate to configuration. Other methods such as system pressure switches, brake handle position indicators, etc., may not provide the warning required by this rule.

AC 29.1157. § 29.1157 CARBURETOR AIR TEMPERATURE CONTROLS.

a. Explanation.

(1) This section addresses the air temperature control for carburetor equipped reciprocating engines.

(2) For rotorcraft which have more than one such engine installed, a separate carburetor air temperature control must be provided for each engine.

b. Procedure.

(1) The engine air induction system should incorporate a means for the prevention and elimination of ice accumulations by preheating the air prior to its entry into the carburetor.

(2) Manually operated push/pull systems have been used which operate a flapper valve inside the air induction system. One such system for each engine is one method of compliance.

AC 29.1159. § 29.1159 SUPERCHARGER CONTROLS.

a. Explanation.

(1) This section addresses the accessibility to supercharger controls in the cockpit, if installed.

(2) These controls must be located so they are easily reached by the pilots or, if the rotorcraft is so configured, by a flight engineer.

b. Procedure.

(1) The location and shape of the controls should be conveniently accessible and sufficiently unique to preclude inadvertent actuation of the wrong control.

(2) Compliance is typically shown by a cockpit evaluation.

AC 29.1163. § 29.1163 (Amendment 29-26) POWERPLANT ACCESSORIES.

a. Explanation.

(1) This section addresses the interface requirements for powerplant accessories which are mounted on the engine or rotor drive system components.

(2) Areas which should be addressed include structural loads imposed upon the engine case and isolation between the accessory and engine oil systems. Electrical equipment isolation from flammable fluids or vapors should be addressed as well as the effect of an accessory failure on the continued operation of the engine and drive system components.

b. Procedures.

(1) Accessories installed and certified by the engine manufacturer can be mounted on the engine without additional justification.

(2) Any accessory to be mounted on the engine, which was not certificated with the engine and does not meet the engine installation design manual requirements, should have a structural analysis showing the mounting of that accessory on the engine will not induce loads into the engine case which are higher than the original design loads.

(3) When the accessory is mounted and operating on the engine, it should not be possible to contaminate either the engine or accessory oil systems. This contamination can take the form of debris following a failure, airborne dirt or water, or any other substance that would impair proper operation of the engine or accessory. Compliance with these requirements can be accomplished by a combination of test and analysis. The design interface should be such that when the equipment is operating, there are no high/low pressure differentials between the components which would induce fluid transfer between components resulting in a low fluid level in one component and an overfill condition in the other component. Where this potential exists, an analysis and/or test should be used to demonstrate compliance.

(4) Engine mounted accessories which are subject to arcing and sparking must be isolated from all flammable fluids or vapors to minimize the probability of fire. This can be accomplished by isolating the electrical equipment from the flammable fumes or vapors or by isolating the flammable fumes or vapors from the potential ignition source. Compliance can be shown by analysis.

(5) A failure mode and effect analysis should be submitted which shows that a failure of any engine mounted and driven accessory will not interfere with the continued operation of the engine. If a hazard is created by the continued rotation of an engine driven accessory after a failure or malfunction, provisions to stop its rotation or eliminate

the hazard must be provided. The effectiveness of this device should be demonstrated by test.

(6) The main transmission and rotor drive system should be protected from excessive torque loads and damage imposed upon them by accessory drives. One method which has been used is a torque limiting device (i.e., shear section of main rotor drive shaft). The effectiveness of any protection device should be demonstrated by test.

AC 29.1165. § 29.1165 (Amendment 29–12) ENGINE IGNITION SYSTEMS.

a. Explanation.

(1) This section defines the design requirements for battery, generator, and magneto ignition systems installed in either reciprocating or turbine engine powered rotorcraft.

(2) The requirements specify common failure modes of batteries, generators, and installed wiring which must be considered in the design process and provides for crew warning of malfunctions.

b. Procedures.

(1) In a battery ignition system, a generator should be available to supply current to the engine ignition system if the battery fails. The generator power should be switched over automatically with an appropriate warning to the crew. The automatic switchover can be accomplished by a low voltage sensor which activates a relay that simultaneously activates a caution light in the cockpit.

(2) An electrical load analysis should be conducted to insure that the capacity of the batteries and generator is large enough to meet the worst-case demands in the system. If there are other electrical system components installed which draw from the same source, the analysis should show that there is sufficient electrical power available from either the battery or the generator to operate all components simultaneously.

(3) The requirements of § 29.1165(c)(1) through (3), should be demonstrated by test. A proposed test plan should be coordinated with the FAA/AUTHORITY prior to conducting the testing.

(4) Compliance with the requirements of § 29.1165(d) can be shown by a failure mode and effect analysis.

(5) The requirements of § 29.1165(e) and (f) are self-explanatory.

SUBPART E - POWERPLANT

POWERPLANT FIRE PROTECTION

AC 29.1181. § 29.1181 (Amendment 29–26) DESIGNATED FIRE ZONES: REGIONS INCLUDED.

a. Explanation. A designated fire zone is a zone on a rotorcraft within which it is assumed (based on past operational experience) that a severe fire (see definitions) will occur sometime in the service life of each rotorcraft; therefore, proper protection must be provided for each new or modified unit by meeting the requirements of §§ 29.1183 through 29.1203. Some common examples of designated fire zones are:

(1) For reciprocating engines:

(i) The power section.

(ii) The accessory section.

(iii) The complete powerplant compartment, if there is no isolation between the power and accessory sections.

(2) Any auxiliary power unit (APU) compartment.

(3) Any fuel burning heater or other combustion equipment installation described under § 29.859.

(4) For Turbine Engines:

(i) The compressor section.

(ii) The accessory section.

(iii) The combustor turbine and tailpipe section unless they--

(A) Do not contain lines and components carrying flammable fluids or gases; and

(B) Are isolated from the designated fire zone prescribed in § 29.1181(a)(6) by a firewall that meets § 29.1191.

(5) Any other essential or non-essential device or system (such as spray rigs using flammable fluids) capable of leaking flammable fluid or gas and creating a severe fire.

b. Definition. Severe fire. See definition in paragraph AC 29.859.

c. Procedures. A FAA/AUTHORITY/applicant design review should be conducted early during certification to identify all designated fire zones and to define the detailed method-of-compliance to be used to meet the requirements of §§ 29.1183 through 29.1203. If significant design changes are made the design change and the method-of-compliance should be re-reviewed to insure they properly support the certification requirements.

AC 29.1183. § 29.1183 (Amendment 29-22) LINES, FITTINGS, AND COMPONENTS.

a. Explanation. This section requires that any line, fitting or other component of a flammable fluid, fuel or flammable gas system which carries, conveys or contains the fluid or gas in any area subject to engine fire conditions (i.e., a severe fire) must be at least fire resistant (reference § 1.1 for definition of fire resistant and see paragraph AC 29.859 which defines a severe fire). An exception is for flammable fluid tanks and supports which are part of and attached to the engine or are in a designated fire zone. These items are required to either be fireproof (see § 1.1 for definition of fireproof and see paragraph AC 29.859 which defines a severe fire) or to be enclosed by a fireproof shield, unless fire damage to any non-fireproof part (e.g., secondary line or valve support) will not cause leakage of a flammable gas, flammable fluid or otherwise prevent continued safe flight and landing of the rotorcraft. All such components must be shielded, located, otherwise protected, or a combination to safeguard against the ignition of leaking flammable fluids or gases. Integral oil sumps of less than 25 quarts capacity on a reciprocating engine need not be fireproof or enclosed by a fireproof shield; however, they should be fire resistant. Most integral sumps in this category are, by natural design and material selection, fire resistant. Exemptions to the preceding requirements are as follows:

(1) Lines, fittings and components already approved under Part 33 as part of the engine itself;

(2) Vent and drain lines (and their fittings) whose failure will not result in or add to an operational fire hazard. In addition, all flammable fluid drains and vents must discharge clear of the induction system air inlet and other obvious ignition hazards.

b. Procedures. A detailed review of the design should be conducted to identify and quantify all lines, fittings, and other components which carry flammable fluids and/or gases and are in areas subject to engine fire conditions such as engine compartments and other fire zones. Once these items are identified the design means of fire protection should be selected and validated, as necessary, during certification. For materials and devices that cannot be qualified as fireproof or fire resistant by similarity or by known material standards, testing to severe fire conditions (see definition, AC 20-135, and AC 23-2 for detailed requirements) should be conducted on full-scale specimens or representative samples to establish their fireproof or fire resistance capabilities. Exceptions to these standards (as provided in the regulatory section) should be reviewed and approved/disapproved on a case-by-case basis during certification. Also, operational fire hazards from drains, vents, and other similar sources should be identified and eliminated during certification.

AC 29.1185. § 29.1185 FLAMMABLE FLUIDS.

a. Explanation. This section requires that fuel, flammable fluid or vapor tanks, reservoirs or collectors be sufficiently isolated from engines, engine compartments, and other designated fire zones so that hazardous heat transfer from these areas to fuel, flammable fluid, and vapor tanks, reservoirs or collectors is prevented in either normal or emergency service.

b. Definitions.

(1) Fuel or Flammable Fluid Collector. Any device such as a large valve, accumulator, or pump that contains a significant amount of flammable fluid, fuel, or vapor (e.g., the volume equal to 10 ounces or more of fluid).

(2) Flammable Fluid or Vapor Tank. Any fuel, flammable fluid or vapor tank, reservoir or collector.

(3) Sufficiently Isolated. Fuel, flammable fluids, or vapors in a tank, reservoir, or collector are insulated, removed, otherwise protected or a combination such that their worst case temperatures (the worst case measured or calculated surface temperature of their containers) in either normal or emergency service is always 50° F or more away from the autoignition temperature of the fuel, flammable fluid, or vapor in question.

(4) Minimum Autoignition Temperature. The temperature at a given vapor pressure at or above which liquid fuel or fuel vapor will self combust. When determining the minimum design value of autoignition temperature which will occur in either normal or emergency operations, the critical, in-service combination of vapor pressure and fuel temperature should first be determined.

(5) Hazardous Heat Transfer. A total incident heat flux (a combination of conduction, convection, and radiation, as applicable) from or in an engine compartment or any other designated fire zone which would raise the temperature level of a flammable fluid or fuel, their vapors, or the surface temperature of their containers to within 50° F or less of the minimum in-service autoignition temperature. Typically, the most critical heat transfer case to be considered is emergency service where a severe fire (see definition) is assumed to occur in each engine compartment and each designated fire zone on a case-by-case basis.

(6) Severe Fire. See definition in paragraph AC 29.859.

c. Procedures.

(1) The fuel, flammable fluid, and vapor system designs should be reviewed early in the certification process to insure that all fuel or flammable fluid or vapor tanks are properly identified and isolated from engines, engine compartments, and other designated fire zones during both normal and emergency operations such as in-flight engine compartment or other fire zone fires. In some cases fuel or flammable fluid components must be located in an engine compartment or other designated fire zone. In these cases, an equivalent safety finding (which considers the design, construction, materials, fuel lines, fittings, and controls used in the system, or system segment, contained in the engine compartment or other designated fire zone) should be undertaken as a part of the normal certification process. If the level of safety provided is equivalent to that provided by removing the system or system segment from the engine compartment or designated fire zone, then the design should be accepted. For fuel tanks only, isolation is required by regulation to be achieved by use of either a firewall (reference paragraph AC 29.1191 for Firewall Requirements) or by use of a shroud. A shroud if used should be fireproof (see § 1.1 for definition and the definition of a Severe Fire for further details) and should be drainable (or otherwise inspectable) to insure the fuel tank is not leaking in service. For other flammable fluid or vapor tanks, the regulations allow either the identical treatment previously described for fuel tanks (i.e., firewalls or shrouds) or, alternatively, use of an equivalent safety finding. Regulations require that the equivalent safety finding be based on system design, tank materials, tank supports, and flammable fluid system connectors, lines, and controls. In all cases the flammable fluids, fuels, and vapors should be sufficiently isolated from hazardous heat fluxes during both normal and emergency operations to prevent autoignition.

(2) In addition, the regulations require at least ½-inch of clear airspace between each flammable fluid or vapor tank, and each firewall or shroud that isolates the system, unless equivalent means (such as fireproof insulation) are used to prevent hazardous heat transfer from each engine compartment or other fire zone to the flammable fluid or vapor mass (or its container surface) at the fluid or vapor's minimum autoignition temperature. If in-service structural deflections are significant, they must be taken into account when certifying the ½-inch minimum clear airspace requirement. For example, if a ½-inch clearance exists on the ground but in some normal and emergency flight conditions (e.g., autorotation) the ½ inch is reduced to ¼ inch at a critical time (in-flight engine fire), then the design (static) configuration should have at least a ½ plus ¼ equals 3/4-inch static clear airspace to insure the regulation's intent is met. Alternatively, fireproof insulation or additional stiffeners could be used to insure the regulation's intent is met (i.e., the thermal equivalent of ½ clearance is maintained at all times). Any material used as insulation on or used adjacent to flammable fluid or vapor tank, should be certified as chemically compatible with the flammable fluid or vapor and to be non-absorbent in case of fuel or vapor leaks. Otherwise, the material should either be treated for compatibility and non-absorbency or not accepted.

AC 29.1187. § 29.1187 DRAINAGE AND VENTILATION OF FIRE ZONES.

a. Explanation. To insure that any component malfunction which results in fuel, flammable fluid or vapor leaks is safely drained or vented overboard and to insure that a fire hazard is not created during either normal or emergency service, there should be complete, rapid drainage and ventilation capability present for each part of the rotorcraft powerplant installation and any other designated fire zone which utilizes flammable fluid or vapor carrying components. As a minimum, the routing, drainage, and ventilation system should accomplish the following:

(1) It should be effective under normal and emergency operating conditions.

(2) It should be designed and arranged so that no discharged fluid or vapor will create a fire hazard under normal and emergency operating conditions.

(3) It should prevent accumulation of hazardous fluids and vapors in any engine compartments and other designated fire zones.

b. Definitions. Drip Fence. A physical barrier that interrupts the flow of a liquid on the underside of a surface, such as a fuel tank, and allows any leaked liquid to drip from the surface away from a hazardous locations to a safe external drain.

c. Procedures. The design of flammable fluid and gas systems running through engine compartments and other designated fire zones should have a thorough hazard analysis performed early during certification. The analysis should be updated periodically as design changes dictate. The hazard analysis should identify and quantify all normal and emergency service failures that could result in leakage of fuel, flammable fluids and vapors. Once these potential hazards are identified and quantified, appropriate design features, such as drains, drip fences and vents, that minimize or eliminate the hazard should be provided. These means should be analyzed and/or tested, as necessary, to insure that their size, flow capacity, and other design parameters are adequate to rapidly remove hazardous fluids and vapors safely away from the rotorcraft under normal and emergency flight conditions. Typically a venting or draining system should be designed to a 3-to-1 flow capacity margin over the probable worst case leak to which it could be subjected. Adverse effects such as clogging and surface tension flow reduction should be accounted for in design. Testing, including flight testing, using inert fluids or vapors may be necessary for proper design certification. In some instances it may be appropriate to include ventilation and drainage tests when the aircraft is parked.

AC 29.1189. § 29.1189 (Amendment 29-26) SHUTOFF MEANS.

a. Explanation.

(1) This section establishes the requirements for controlling hazardous quantities of flammable fluids which flow into, within, or through designated fire zones.

(2) When any shutoff valve is operated, any equipment, including a remaining engine, which is essential for continued flight, cannot be affected.

b. Procedures.

(1) Combustible fluid supply lines which pass into, within, or through a firewall into the fire zone must incorporate shutoff valves. This requirement does not apply to lines, fittings, and components which were certified with and are part of the engine. These requirements do not apply to oil systems for Category B rotorcraft with reciprocating engines with less than 500 cubic inches displacement or to any other installation where all components, including the oil tanks, are fireproof or are located in an area that will not be affected by an engine fire.

(2) Eight fluid ounces or less of a combustible fluid is not considered hazardous and no more than this amount should be present after activating the shutoff valve.

(3) Engine isolation is to be maintained when incorporating shutoff valves into engine fuel and lubrication lines. The design must insure that when one engine is shut down or fails and the fuel and lubrication fluid shutoff valves are activated, the remaining good engine is not affected in any way, and the rotorcraft can continue safe flight to a landing. This should be demonstrated by test.

(4) Each shutoff valve located in a fire zone should be fireproof. If the shutoff valve is located outside of the fire zone, then it should be at least fire resistant or protected so that it will function under a worst case fire condition within a fire zone. This should be demonstrated by test.

(5) Except for ground-use-only auxiliary power unit installations, the flammable fluid shutoff to all engine installations must be protected from inadvertent operation. Where electrical shutoffs are used, the switches must be guarded or require double actions. If the shutoffs are mechanically activated, the design of the knob and the location of the lever must be such that inadvertent actuation cannot occur. It must be possible to reopen the shutoff valve in flight after it has been closed and this should be demonstrated by test.

AC 29.1191. § 29.1191 (Amendment 29-3) FIREWALLS.

a. Explanation. This section states the certification requirements for the proper certification of fireproof protective devices such as firewalls, shrouds, or equivalent. These devices are necessary to isolate each engine (including combustor, turbine, and tailpipe sections of turbine engines and auxiliary propulsion units (APU); each APU; each combustion heater; each unit of combustion equipment; or each high temperature device (or source) from personnel compartments and critical components (not already protected under § 29.1191). The isolation of these fire zones is necessary to prevent the spread of fire, prevent or minimize thermal injuries and fatalities, and prevent damage to critical components that are essential to a controlled landing. Even though § 29.1191(b) implicitly excludes APU's, combustion heaters, and other combustion equipment that are not used in flight; they should be protected by fireproof enclosures, because of § 29.901(d) and the requirements of the relevant parts of §§ 29.1183 through 29.1203. This is because, even if the device is rendered inoperative in flight, it typically contains residual heat, fuel, fumes and potential ignition sources (i.e., "potential hazards"). Each fireproof protective device must, by regulation, meet the following criteria:

(1) Its design and location must take into account the probable fire path from each fire zone or source considering factors such as internal airflow, external air flow, and gravity.

(2) It must be constructed so that no hazardous quantity of air, fumes, fluids, or flame can propagate through it to unprotected parts of the rotorcraft.

(3) Its openings (e.g., shaftholes, lineholes, etc.) must be sealed with close fitting fireproof grommets, bushings, bearings, firewall fittings, or equivalent that prevent burn through and leakage of hazardous fumes or fluids from the fire zone.

(4) It must be fireproof (see definition).

(5) It must be either corrosion resistant or otherwise safely protected from corrosion.

b. <u>Definitions</u>.

(1) <u>Fireproof Protective Device</u>. A fireproof protective device is a device such as a firewall, shroud, enclosure, or equivalent used to isolate a heat or potential fire source (severe fire) from personnel compartments and from critical aircraft components which are essential for a controlled landing.

(2) <u>Fireproof</u>. Fireproof is defined in § 1.1 "General Definitions."

(3) <u>Controlled Landing</u>. A landing which is survivable (i.e., does not fatally injure all occupants) but may produce an unairworthy, partially salvageable, or unsalvageable rotorcraft.

(4) <u>Severe Fire</u>. See definition in paragraph AC 29.859.

c. <u>Procedures</u>. Fireproof protective devices are typically certified by analysis, tests, or a combination conducted during the certification process, including flight tests or simulated flight tests, as follows:

(1) Fireproof protective devices should be provided wherever a hazard exists which requires isolation from a severe fire (see definition) to avoid fires in personnel compartments and to avoid thermal damage to critical components (such as structural elements, controls, rotor mechanisms, and system components) that are necessary for a controlled landing. A thorough hazard analysis should be conducted during certification to identify, define and quantify in order of severity (i.e., maximum temperature, hot exposed area, etc.) all thermal hazards or zones that require fireproof protection in a given design. Engines (including the combustor, turbine, and tailpipe sections of turbine engines), APU's, combustion heaters, and combustion devices are required by regulation to be isolated. Other high temperature devices may also require isolation because of local hot spots (which occur during normal operations or from failure modes) that can thermally injure occupants or cause spontaneous combustion of surroundings. A hazard analysis should identify these potential problems and provide proper certification solutions.

(2) Fireproof protective devices should be able to withstand at least 2000 ± 150° F for at least 15 minutes (reference AC 20-135). The fireproof protective device should allow the protected parts, subsystems or systems to perform their intended function for the duration of a severe fire (see definition). For firewalls, examples of flat, geometry materials undergoing uniform heat fluxes with material gauges that automatically meet the certification requirements are given in figure AC 29.1191–1. If firewalls are utilized that involve other materials, significant geometric changes, or significantly non-uniform heat fluxes, then automatic compliance may not be assured. In such cases the fireproof protective devices should be analyzed and, in some cases, tested in accordance with AC 23–2 to ensure proper certification. For example, a curved protective surface may absorb a uniform incident heat flux unevenly and create a local hot spot that exceeds 2050° F that burns through in less than 15 minutes; whereas, a flat surface of equal thickness would not exceed 2050° F and would not burn through in less than 15 minutes. It should be noted that composite materials are not generally used for protective devices because of their inability to withstand high temperatures (i.e., exceedance of the glass transition temperature); however, some specially formulated composites have been previously certified as engine cowlings. Titanium is an acceptable material for fireproof protective devices such as firewalls. However, use of titanium should always be carefully considered and reviewed, because it can lose all structural ability and burn severely (self combust) above 1,050° F, under certain thermodynamic environments, and contribute to the fire instead of providing the intended fire protection. AC 33–4, "Design Considerations Concerning the Use of Titanium in Aircraft Turbine Engines" and MIL-HDBK-5D contain more detailed information on the unique thermal properties of titanium.

FIGURE AC 29.1191-1
TABLE OF MATERIALS AND GAGES ACCEPTABLE
FOR FIREPROOF PROTECTIVE DEVICES WITH FLAT
SURFACE GEOMETRIES [1]

MATERIAL[2]	MINIMUM THICKNESS[3]
Titanium Sheet	.016 in
Stainless Steel	.015 in
Mild Carbon Steel	.018 in
Terne Plate	.018 in
Monel Metal	.018 in
Firewall Fittings (Steel or Copper Base)	.018 in[4]

NOTES:

(1) Assumes essentially flat vertical or horizontal surfaces undergoing a uniform heat flux. Any significant variation in either geometry or heat flux distribution should be examined in detail for adequate gauge thicknesses on a case-by-case basis.

(2) Must have corrosion protection if not inherent in the material itself.

(3) The minimum thickness is for thermal containment only. Structural integrity considerations may require thickness increases. MIL-HDBK-5D contains material allowable versus temperature data for common metallic materials.

(4) This is the minimum wall thickness measured at the smallest dimension (e.g., thread root or other location) of the part.

(5) Distortion of thin sheet materials and the subsequent gapping at lap joints or between rivets is difficult to predict; therefore, testing of the simulated installation is necessary to prove the integrity of the design. However, rivet pitches of 2 inches or less on non load-carrying titanium firewalls of .020 inch or steel firewalls of .018 inch are acceptable without further testing.

(3) The probable path of a fire (as affected by internal and external air flow during normal flight and autorotation, gravity, flame propagation paths, or other considerations) should be taken into account when performing the hazard analysis of item (1). Such a review will insure that fireproof protective devices are placed in the proper location for intercepting, blocking or containing a severe fire before occupants are injured and a controlled landing is prevented. If the probable path cannot be readily determined by inspection or analysis, testing using simulated airflows, rotorcraft attitudes, and dyed inert fluids or vapors can be used to aid in this determination.

(4) Each opening in a protective device should be sealed with close fitting sealing devices such as fireproof grommets, bushings, firewall fittings, rotating seals or equivalent that are at least as effective as the fireproof protective device itself. This is necessary to insure that no local breakdowns in protection occur. For materials not listed as acceptable in item (1), FAA/AUTHORITY standards and analysis and testing should be required in accordance with the definition of a severe fire for proper substantiation.

(5) Each protective device should be fireproof in order to withstand a severe fire (see definition). Unless designs and materials have been previously FAA/AUTHORITY approved (e.g., see Item 1), the protective device's design and material selection should be tested to insure its fireproof thermal and structural integrity. A full-scale test of a structurally loaded article or a representative sample should be conducted to insure proper compliance is achieved. Also, the continued sealing ability of the protective device in its deformed state due to a hard controlled landing should be considered during certification (e.g., use of ductile materials). The corrosion environment should be defined and appropriate protection provided. Phased inspections should be specified, if necessary, to insure continued corrosion integrity. Certification tests for adequacy of corrosion protection should be conducted, using sample plates or by other equivalent means, as required.

AC 29.1193. § 29.1193 (Amendment 29-13) COWLING AND ENGINE COMPARTMENT COVERING.

a. Explanation.

(1) Section 29.1193(a) requires the cowling and engine compartment coverings to withstand structural loads experienced in flight.

(2) In order to prevent pooling of flammable fluids, § 29.1193(b) requires ventilation and complete drainage from the cowling and engine compartment as specified in § 29.1187.

(3) In § 29.1193(c), (d), and (e), clarification of fireproof requirements is provided along with interaction between the requirements of § 29.1191 for firewalls.

b. Procedures.

(1) Compliance with § 29.1193(a) can be shown by analyzing the cowling and engine compartment covering and determining that no structural degradation will occur under the highest loads experienced on the ground or in flight.

(2) Compliance with § 29.1193(b) can be accomplished by ensuring that the drain will discharge positively with no traps and is a minimum of 0.25 inches in diameter. No drain may discharge where it might cause a fire hazard. This can be demonstrated by colored liquid flowing through the drain system while in flight. The dye should not impinge on any ignition source during any approved flight regime.

(3) Compliance with the fireproof requirements of § 29.1193(c), (d), and (e) can be accomplished by demonstrating that the material will withstand a 2,000° F ± 50° F flame for 15 minutes while still fulfilling its design purpose. This testing should accurately simulate, as near as practicable, the likely fire environment to prove the materials and components will provide the necessary fire containment when exposed to a fire situation in service. In addition to the fireproof requirements, the requirements of § 29.1191 must also be met. The primary objectives are:

(i) To contain and isolate a fire and prevent other sources of fuel and/or oxygen from feeding the existing fire; and

(ii) To ensure that components of the engine control system will function effectively to permit a safe landing and/or shutdown of the engine.

AC 29.1193A. § 29.1193 (Amendment 29-26) COWLING AND ENGINE COMPARTMENT COVERING.

a. Explanation. Amendment 29-26 adds a new § 29.1193(f) that requires redundant retention means for each panel, cowling, engine, or rotor drive system covering that can be opened or readily removed. Conventional fasteners for these devices are subject to frequent operation by maintenance personnel and have deteriorated, failed from wear or vibration, or been left unsecured after preflight inspections. Such a failure could be hazardous if a loose panel, cowling, or covering strikes, or is struck by, the rotors or by critical controls.

b. Procedures.

(1) Compliance with § 29.1193(f) can be accomplished by simulating, or actually failing, one or more of the retention devices or by structural analysis. If a failure of a single retention device can contribute to multiple failures, these multiple failures should be considered. It should be shown that the cowling or cover will not open, strike, or be struck by the rotor or other critical component.

(2) Consideration should be given to minimize the possibility of latches being improperly closed that could result in a cowl coming open in flight.

(3) The failure of one latching device should not cause the failure of another latching device.

(4) The consequences of “forgetting” to latch a cowl should be considered.

(5) The use of safety straps should be considered to minimize the impact of a latching device failure.

AC 29.1194. § 29.1194 (Amendment 29-3) OTHER SURFACES.

a. Explanation. This section states the fire resistance requirements for material surfaces near engine compartments and designated fire zones (other than tail surfaces not subject to heat, flames, or sparks emanating from a designated fire zone or engine compartment).

b. Definition.

(1) Other Surface. Any airframe, system, or powerplant component aft of and near an engine compartment, a designated fire zone, or another heat source which would receive a heat flux as a result of a fire in the engine compartment or fire zone that would require the component to be fire resistant.

(2) Fire Resistant. In accordance with § 1.1, is defined as follows:

(i) Sheet metal or structural members with the capacity to withstand the heat associated with the fire at least as well as aluminum alloy in dimensions appropriate for the purpose for which they are used.

(ii) Fluid carrying lines, fluid system parts, wiring, air ducts, fittings and powerplant controls with the capacity to perform their intended functions under the heat and other conditions resulting from a fire.

(3) Fire. A fire in either an engine compartment or a designated fire zone is assumed to occur that produces a heat flux on a system, airframe or powerplant component aft of or near the fire. The effect of each such fire on other surfaces must be considered on a case-by-case basis to determine the critical case. Unless a more rationale definition is furnished and approved during certification, the fire in any engine compartment or designated fire zone should be assumed, for purposes of analysis, to be a severe fire (see definition in paragraph AC 29.859).

c. Procedures.

(1) Other surfaces should be identified during certification by a design review and by a conservative, thorough hazard analysis based on an analytical estimate of the total heat flux (i.e., conduction, convection, and radiation in combination, as applicable) using the definition of a severe fire and of the resultant “other surface” temperature based on a single fire occurring in each engine compartment and designated fire zone, on a case-by-case basis. Once the other surfaces are identified and their severe fire induced maximum temperatures determined, their configuration and material selection should be reviewed on a case-by-case basis to determine either that they are fire resistant, that they can be made fire resistant (within the limits of practicability), or that it is impracticable to make them fire resistant. If the non-fire resistant other surfaces can be readily made fire resistant they should be. If it is impracticable to make them fire resistant, then they should be relocated, insulated, or a combination in order to reduce the total incident heat flux (and, thus, lower their surface temperature) so that they no longer need to be fire resistant. If insulation is used to shield a surface that is subjected to a significant temperature, it must be fire resistant.

(2) A partial validation of analytical heat flux models using the definition of a severe fire can sometimes be achieved during certification tests by using thermocouples or heat-sensitive stickers to measure in-flight temperature ranges and distributions on other surfaces from known thermal environments in engine compartments or other designated fire zones.

AC 29.1195. § 29.1195 (Amendment 29-17) FIRE EXTINGUISHING SYSTEMS.

a. Explanation. This section specifies the types of rotorcraft which must have fire extinguishing systems and the number of discharges. The types of tests and airflow conditions are also specified for demonstration of compliance.

b. Procedures.

(1) The requirements are applicable to each turbine engine powered rotorcraft, Category A reciprocating engine powered rotorcraft, and each Category B reciprocating engine powered rotorcraft with an engine of more than 1,500 cubic inches. There must be a fire extinguishing system for the designated fire zones defined in § 29.1181.

(2) A fire extinguishing system should dilute all of the atmosphere within and entering a compartment with sufficient inert agent that it will not support combustion, and continue the process long enough to extinguish the existing flame and either dissipate the vapors or eliminate the ignition sources. Conventional systems utilize perforated tubing or discharge nozzles to distribute a specific quantity of agent in approximately 2 seconds. HRD (high rate of discharge) systems utilize open end tubes to deliver a given quantity of agent within 1.35 seconds for CO_2 and 1 second for all other agents. The HRD systems are recommended for use in compartments having high airflow where the required discharge rates can be more effectively provided by a HRD rather than a perforated tubing system. Tests indicate that unrestricted release through such an open end tube distribution system can be relied on for adequate

distribution, provided the outlets are located properly. Although the discharge times given above are considered satisfactory, any reduction in discharge time below that specified would improve system effectiveness. However, consideration should be given to the time requirements for draining accumulated combustibles, dissipating combustible vapors and cooling or eliminating ignition sources to assure that the minimum agent concentration is maintained for a duration sufficient to prevent reignition of the combustibles.

(3) The possible variety of tankage and plumbing configurations to accomplish the result should be examined for each specific aircraft in order to achieve the optimum. Systems can vary from tankage in a central location, which is directed through complex distribution systems to various hazards, to agent which is tanked adjacent to each hazard. Terminology generally accepted to define various arrangements is as follows:

(i) Central System: A single supply of agent, centrally located, with valves to direct the agent to any protected zone or zones.

(ii) Individual System: A separate supply of agent for each protected zone or zones.

(4) The selection of the distribution system should be made with full cognizance of the hazards to be covered. The distributor system (i.e., discharge nozzles fed individually by lines from a central manifold) is the most efficient. The complexity of such a system, however, may show it in a less favorable light than the loop or ring system (i.e., orifices drilled in a distribution line, the loop being fed from one end, and the ring being fed from a point on a continuous circle) as far as weight, complexity of manufacture and types of hazard to be covered are concerned. For HRD systems, open feed lines are recommended. In high air flow zones, outlets should be located as far upstream as possible with the discharge directed across the air stream and slightly downstream such that a helical spray pattern is produced. In zones of low or negligible air flow, the outlet location is not critical but a location at the top-center of the zone with the agent directed downward is suggested.

(5) In a conventional CO_2 system, all lines upstream of direction valves should be 4,000 PSI (27,600 kPa) burst and lines which are open should be 2,000 PSI (13,800 kPa) burst. Care should be taken to insure that all valving and/or equipment in the distribution line has an appropriate flow rate. Expansion of fittings, tee, etc., should be checked to insure that not over 150 percent of the inflow area exists downstream. 130 percent is accepted as the safest target value. If overexpansion occurs, snow will form and plug the lines. Because of high storage pressure, the orifice areas of a conventional CO_2 system seem to act as the flow control with system flow losses as a minor effect. Because of this, distribution systems of 50 ft. (15.24m) or less can be satisfactorily computed by the following factor:

(i) Line Area = .10 sq. in./lb CO_2/sec (142.2 mm^2/kgCO_2/sec).

(ii) Orifice Area = .072 sq. in./lb CO_2/sec (102.4 mm^2/kgCO_2/sec) (72 percent of equivalent line area).

(iii) Min. Orifice Size = 1/16 in. (1.6 mm) diameter.

(6) In low pressure systems such as "CB" and CH_3Br, line and fitting losses become a greater effect in the discharge rates and distribution than was true with CO_2. Consideration should be given to the small I.D. of an AN line fitting with respect to the I.D. of the mating tube sizes. This may be done by extra pressure drop allowances, by enlarging these fittings, or by making special fittings. Within reasonable line lengths, however, area factors can be used with fair accuracy. (It is generally conceded that a system designed to these factors, especially a complex layout, should be carefully tested or analyzed for time of discharge and distribution.) These areas are as follows:

(i) Line Area = .07 sq. in./lb agent/sec (99.6 mm^2/kg agent/sec).

(ii) Orifice Area = .05 sq. in./lb agent/sec (71.1 mm^2/kg agent/sec) (72 percent of equivalent line area).

(iii) Min. Orifice Area = 1/32 in. (.8 mm) diameter.

(7) For HRD systems of all types, feed line cross-sectional area is dependent upon the rate desired and upon system volume considerations. The minimum diameter of the feed line is established by the required rate; the maximum diameter of the feed line, and by the need for keeping the system volume at a minimum. Specifically, with the propelling gas in a system pressurized to 400 PSI (2760 kPa), the "volumetric efficiency" should be at least 0.50; that is, the original volume of the propelling gas in the system should be at least ½ the volume of the entire system, including that of the agent container. It is recommended that for HRD systems the feed lines be open. No nozzles or series of perforations are required. It is believed that the unrestricted release of the more volatile liquid agents, as well as carbon dioxide, can be relied upon for adequate distribution, provided the outlets are properly located. It is important that any such system be carefully tested for time of discharge, distribution, and minimum concentrations.

(8) From the basic definition, the system should be effective if the distribution of the agent floods the various portions of a compartment simultaneously and dilutes the incoming air. It is noted that the typical high flow compartment requires a greater proportion of its total agent discharged at the air inlet than does the conventional low air flow zone. All parts of the fire extinguisher system directed to any one powerplant installation should be discharged simultaneously. The theory behind the HRD type system is that with rapid discharge of the agent, the concentration necessary for extinguishment is reached more rapidly with correspondingly less time for dissipation or dilution of the agent by incoming air. The duration of this critical concentration necessary for extinguishment is believed to remain the same as for conventional systems.

(9) Detailed system configuration recommendations are not available for conventional systems; however, the recommendations on the configuration of HRD systems would probably apply equally well to all types. For HRD systems, it is recommended that feed lines be as short as possible, requiring that agent containers be as close as practical to the zones to be protected. Feed lines should be direct; the fewer fittings and turns, the better. Expansions and restrictions have adverse effects on rate; and it is probable that in a feed line with long rises or many changes or direction, quantities of propelling gas can get past a liquid agent, thus reducing the discharge rate and making the discharge sporadic and ineffective. Where such fittings, changes of direction, and long vertical rises are unavoidable, compensation in the form of additional agent may be necessary.

(10) A fixed "one shot" fire extinguisher system should be provided for the heater extinguisher system in order to extinguish the fire in the combustion chamber. The regions surrounding the heater and combustion chamber must also be protected if these regions contain components with potential combustible leakage. No fire extinguishment is needed in cabin air passages.

AC 29.1197. § 29.1197 (Amendment 29-13) FIRE EXTINGUISHING AGENTS.

a. Explanation.

(1) Fire extinguishing agents used in rotorcraft fire extinguishing systems must be capable of extinguishing any fire in the area where the system is installed.

(2) The extinguishing agent must maintain its effectiveness after prolonged storage under the environmental conditions of the compartment in which it is stored.

(3) If a toxic extinguishing agent is used, the harmful concentration level of the fluid vapors must be determined and it must be shown that it is not possible for this concentration level to enter into any personnel compartment.

b. Procedures.

(1) The fire extinguishing system should dilute all of the atmosphere within and entering a compartment with sufficient inert agent so that combustion cannot be supported. The extinguishing process should continue for a duration sufficient in length to extinguish any existing flame. When a compartment is to be flooded with agent and there is a source of fresh air entering the compartment, the incoming air should be either shut off prior to the release of the agent or rendered inert by directing extinguishing agent into the air blast (preferably the former) or the quantity of agent should be increased to offset the incoming airflow.

(2) There are a number of extinguishing agents which have been used on rotorcraft in the past. The following list identifies the agent and some advantages and disadvantages of each.

Agents	Advantages	Disadvantages
Carbon Dioxide CO_2	Safest agent to use from the standpoint of toxicity and corrosion hazards.	Mental confusion and suffocation hazard to occupants if sufficient gas is discharged into personnel compartments. CO_2 has an extremely large variation in vapor pressure with temperature which makes it necessary to use stronger (heavier) containers than are required for methyl bromide.
Methyl Bromide CH_3Br	More effective for equal mass than CO_2. Approx. 80 percent of this agent by weight as compared to CO_2 is required. Less variation in vapor pressure than CO_2. Much lower container pressure required resulting in lighter containers. Treated magnesium alloys are satisfactory for use in CH_3Br systems outside of the potential fire zones.	Much more toxic than CO_2. Due to its toxic effects on humans, CH_3Br should not be used as a fire extinguisher agent in areas where harmful time concentrations can enter personnel compartment. Aluminum alloy material should not be used in methyl bromide systems due to serious corrosion and possible spontaneous ignition. Rapidly corrodes aluminum, magnesium, and zinc. Tubing systems should be vented at all times and steps should be taken to free the tubing of residual methyl bromide after each discharge. Containers must be recharged at the extinguisher manufacturer's plant or at a depot by specially trained personnel.
Bromo-chloro-methane ("CB") CH_2BrCl	Low vapor pressure compound - 3 PSIA (20.7 Kpa) at 70° F (21.1°C). One of the more effective agents.	Toxic when burned.
Dibro-modi-	Low vapor pressure compound - 14 PSIA (96.5 Kpa) at 70° F	Very toxic when burned.

Agents	Advantages	Disadvantages
fluoro- methane CF_3Br	(21.1° C). One of the more effective agents. Non-corrosive to aluminum, steel and brass.	
Bromotri fluoro- methane CF_3Br	One of the more effective agents. Low toxicity in natural condition and when burned. Non-corrosive to aluminum, steel and brass.	High vapor pressure compound - 220 PSIA (1517 Kpa) at 70° F (21.1° C). Least toxic of agents in burned condition except for CO_2.
Nitrogen N_2	If a fuel tank inerting system using N_2 is provided, use as extinguishing agent may be considered. N_2 offers cooling not available with CF_3Br.	3 - 4 times quantity and rate of conventional agents required.
Note: The relative effectiveness of the various agents listed above is considerably Influenced by the type of system employed, high rate discharge or conventional; by the method of distribution, open end outlet, nozzle, or spray ring; and by the air flow conditions.		

(3) The extinguishing agent must not be affected by the temperature extremes experienced in the compartments in which they are stored. The agent containers should be either “winterized” for extreme temperature operation or so located in the rotorcraft that they will not be subjected to extreme temperatures. Safe limits for unwinterized carbon dioxide cylinders are approximately 0° F (-18° C) to 140° F (60° C). The cartridge detonators have a variable age-with-temperature limit. Contact should be made with the manufacturer for the latest information available for both installation and storage temperatures.

(4) It must be shown by test that the harmful level of toxic fluid or vapors cannot enter into any personnel compartment due to leakage or activation of the system during normal operation of the rotorcraft in flight or on the ground. The entire fire extinguishing system should be mocked-up or installed in the aircraft down to and including distribution tubing and outlets. The tests should be conducted under actual or simulated cruise conditions. The system should be discharged, and compliance verified by use of an appropriate method for measuring agent concentration.

AC 29.1199. § 29.1199 (Amendment 29–13) EXTINGUISHING AGENT CONTAINERS.

a. Explanation.

(1) This section presents the requirements for fire extinguisher containers. The containers are subjected to high internal pressures for the propulsion of the agent as well as a wide range of external environmental temperatures.

(2) The containers must be adequately protected to preclude any adverse effect on the operation of the system from these external influences.

b. Procedures.

(1) Each extinguishing agent container must have a pressure relief valve which will open at a pressure that is below the burst pressure of the agent container. The pressure relief valve lines must be located and protected so that they cannot be clogged by dirt, ice, or other contaminates. Both the agent container burst pressure and the relief valve opening pressure limits should be verified by test. Agent containers which meet military specification, MIL-C-22284, requirements are acceptable.

(2) The containers should be located so that an indicator is readily visible to determine if the container has discharged or the charging pressure is below operating minimums. The number and size of agent containers should be adequate to obtain the established agent concentration and duration for the intended compartment. It is preferred that the agent supply containers and the flow control valves are not located in a fire zone.

(3) The brackets for mounting the containers and securing the discharge lines should be designed to withstand all loads to which they may be subjected due to recoil during discharge or any other applied load factor.

(4) The agent containers should be protected from extreme temperature excursions which could have an adverse effect upon the operation of the extinguishing system. Safe temperature limits for “unwinterized” carbon dioxide cylinders are approximately 0° F (-18° C) to 140° F (60° C). Safe limits for “CB” and CH_3Br spheres are approximately -65° F (-54° C) to 200° F (93° C). The cartridge detonators have a variable age-with-temperature limit and the manufacturer should be contacted for the latest information on installation and storage temperatures. Location of the container in the aircraft should take these temperature limits into consideration.

AC 29.1201. § 29.1201 FIRE EXTINGUISHING SYSTEM MATERIALS.

a. Explanation.

(1) Many different fire extinguishing agents are available for use in fire extinguishing systems. The choice of extinguishing agent should take into account the chemical reaction (if any) between the extinguishing agent and the materials utilized in the extinguishing system. If there are any incompatibilities, they should not create a hazard by creating volatile or toxic vapors or fumes which could feed a fire or cause injury to passengers, crew, or other personnel.

(2) The fire extinguishing components in an engine compartment must be fireproof to ensure operation in the event of a compartment fire.

b. Procedures.

(1) Compliance with the requirements of § 29.1201(a) can be demonstrated by analysis, test, or a combination of both.

(2) Certification data submitted by the applicant should contain a listing of the chemical ingredients of the extinguishing agent and the other materials in the extinguishing system. These data should also show that the chemical reaction (if any) of these materials, when combined, does not create a hazard.

(3) Where chemical compounds exist and the chemical reaction is not predictable when two different compounds are combined, actual tests may be necessary to determine the hazard potential.

(4) Analysis, test, or a combination of both may be used to demonstrate compliance with the fireproof requirement for all fire extinguishing components located within the engine compartment.

AC 29.1203. § 29.1203 (Amendment 29-40) FIRE DETECTOR SYSTEMS.

a. Explanation.

(1) Fire detection systems are required in turbine engine powered rotorcraft, Category A reciprocating engine powered rotorcraft and each Category B reciprocating engine powered rotorcraft where the engine displacement is greater than 900 cubic inches.

(2) This section specifies material, installation, and some operational requirements for fire detectors to ensure prompt detection of fire in the fire zones and other designated areas.

b. Procedures.

(1) The detector system should be designed for highest reliability to detect a fire and not to give a false alarm. It is desirable that it only responds to a fire and misinterpretation with a lesser hazard should not be possible. Engine overtemperature, harmless exhaust leakage, and bleed air leakage should not be indicated by a fire detector system. A fire detection system should be reserved for a condition requiring immediate measures such as engine shutdown or fire extinguishing. There are three general types of detector-procedure systems that are commonly used:

(i) A manual system utilizes warning lights to alert the pilot who then follows prescribed cockpit procedure as a countermeasure. A manual system is adequate for hazards in which a few seconds are not important.

(ii) There is also a semi-automatic system. Occasionally a rotorcraft becomes so complex that the emergency procedure exceeds reasonable expectations of the pilot. In such cases, psychology should be weighted against complexity, and "panic switches," combining multiple procedure functions, should be provided to simplify the mental demands on the pilot. Speed is gained by such designs for hazards which may need it.

(iii) The detector of an automatic system automatically triggers the appropriate countermeasures and warns the pilot simultaneously. Such a system should be carefully evaluated to assure that the advantages outweigh the disadvantages and potential malfunctions.

(2) Fires, or dangerous fire conditions can be detected by means of various existing techniques. The following is a partial list of available detectors:

(i) Radiation-sensing detectors.

(ii) Rate-of-temperature-rise detectors.

(iii) Overheat detectors.

(iv) Smoke detectors.

(v) CO detectors.

(vi) Combustible mixture detectors.

(vii) Fibre-optic detectors.

(viii) Ultraviolet.

(ix) Observation of crew or passengers.

(3) In many rotorcraft it is desirable to have a detection system which incorporates several of these different types of detectors. Radiation-sensing detectors are most useful where the materials present will burn brightly soon after ignition, such as in the powerplant accessory section. Rate of rise detectors are well-suited to compartments of normally low ambient temperatures and low rates of temperature rise where a fire would produce a high temperature differential and rapid temperature rise. It should be noted that under certain circumstances, where a relatively slow temperature increase occurs over a considerable period of time, a fire can occur without detection by rate of rise detectors. Overheat detectors should be used wherever the hazard is evidenced by temperatures exceeding a predicted, set value. Smoke detectors may be suited to low air flow areas where materials may burn slowly, or smolder. Fibre-optic detectors can be used to visually observe the existence of flame or smoke. The three major detector types used for fast detection of fires are the radiation-sensing, rate-of-rise, and overheat detectors. Radiation-sensing detectors are basically “volume” type which senses flame within a visible space. Overheat-fire detectors can be obtained in either “continuous” or “unit” type.

(4) The detector system should:

(i) Indicate fire within 15 seconds after ignition, and show which engine compartment in which the fire is located.

(ii) Remain on for the duration of the fire.

(iii) Indicate when the fire is out.

(iv) Indicate re-ignition of the fire.

(v) Not by itself precipitate or add to the potential of any other hazards.

(vi) Not cause false warnings under any flight or ground operating condition.

A false fire detector indication could significantly increase crew workload, impair crew efficiency, or reduce safety margins and so is classified as a major failure condition. In consequence, such false fire detector indication should be shown to be improbable based on a probability assessment and service experience of the fire detector system. If the probability of the fire detection system experiencing a false indication cannot be shown to be improbable, a secondary means of determining the validity of the fire indication should be provided.

(5) Additional features of the detection system are as follows:

(i) A means should be incorporated so that operation of the system can be tested from the cockpit.

(ii) Detector units should be of rugged construction, to resist maintenance handling, exposure to fuel, oil, dirt, water, cleaning agent, extreme temperatures, vibration, salt air, fungus, and altitude. Also, they should be light in weight, small, and compact, and readily adaptable to desired positions of mounting.

(iii) The detector system should operate on the rotorcraft electric system without inverters. The circuit should require minimum current unless indicating a fire or unless a monitoring system is in use.

(iv) Fixed temperature fire detectors should preferably be set at 100° F (37.7° C) to 150° F (65.6° C) above maximum safe ambient temperature, or higher when in compartments where extremely high rate of rise is normally encountered.

(v) Detector system components located within fire zones should be fire resistant.

(vi) Each detector system should actuate a light which indicates the location of the fire. If fire warning lights are used, they must be in the pilot's normal field of view.

(vii) Two or more engines should not be dependent upon any one detector circuit. The installation of common zone detection equipment prevents the detection system from distinguishing between the engine installations, necessitating shutting down more than one engine.

(6) The sensing portion of the fire detection system should not extend outside of the coverage area into another fire zone. Detectors, with the exception of radiation–sensing detectors, should be located at points where the ventilation air leaves compartments. If a reverse-flow cooling system is used, detectors should be installed at locations which are outlets under both flight and ground operating conditions. Stagnant air spaces should be avoided and the number of ventilation air exits should be kept to a minimum. The ventilation requirements of § 29.1187(e) must also be taken into consideration. Compliance with these recommendations allows the effective placement of a minimum amount of detectors, and still ensures prompt detection of fire in those zones. Radiation-sensing detectors should be located such that any flame within the compartment is immediately sensed. This may or may not be where the ventilation air leaves the compartment.

(7) Fire detectors must be installed in designated fire zones, the combustor, turbine and tailpipe sections of turbine installations.

(i) Engine Power Section (Combustor, Turbine and Tailpipe): This zone is usually characterized by predictable hazard areas which facilitate proper detector location. It is recommended that coverage be provided for any ventilating air outlet as well as intermediate stations where leaking combustibles may be expected.

(ii) Compressor Compartment: This is usually a zone of relatively low air flow velocities, but wide geographical possibility for fires. When fire detectors other than radiation-sensing detectors are used, detection at air outlets provides the best protection, and intermediate detector locations are of value only when specific hazards are anticipated.

(iii) Accessory Bullet Nose: Where such a compartment is so equipped that it is a possible fire zone, its narrow confines permit sufficient coverage with one or more detectors at the outlets.

(iv) Heater Detector Location: An overheat detector should be placed in the hot air duct downstream of the heater. If the heater fuel system or exhaust system configuration is such that it is a fire hazard, the compartment surrounding the heater should also be examined as a possible fire zone.

(v) Auxiliary Power Unit Detector Location: The use of a combustion-driven auxiliary power unit creates another set of typical engine compartments defined and treated as above. Some units are so shrouded with fireproof material that these compartments exist only within the confines of the shroud. They are still, however, fire zones and must have a detection system.